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Feasibility Study of a Rotorcraft Health and Usage Monitoring System (HUMS): Usage and Structural Life Monitoring Evaluation

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Abstract

The objective of this study was to evaluate two techniques, Flight Condition Recognition (FCR) and Flight Load Synthesis (FLS), for usage monitoring and assess the potential benefits of extending the retirement intervals of life-limited components, thus reducing the operator's maintenance and replacement costs. Both techniques involve indirect determination of loads using measured flight parameters and subsequent fatigue analysis to calculate the life expended on the life-limited components. To assess the potential benefit of usage monitoring, the two usage techniques were compared to current methods of component retirement. In addition, comparisons were made with direct load measurements to assess the accuracy of the two techniques.

The data that was used for the evaluation of the usage monitoring techniques was collected under an independent HUMS Flight trial program, using a commercially available HUMS and data recording system. The usage data collect from the HUMS trial aircraft was analyzed off-line using PC-based software that included the FCR and FLS techniques. In the future, if the technique prove feasible, usage monitoring would be incorporated into the onboard HUMS. The benefit of usage monitoring was identified under work accomplished during the first phase of this activity. The results from the operator's perspective is presented in the report NASA CR198446 (ARL-CR-289; DOT/FAA/AR-95/50).

For the selected dynamic components analyzed, the results of the evaluation of the FCR and FLS techniques indicate a potential for extending retirement lives. This is due to the damage accumulation rate for the FCR and FLS techniques being slower ("slow clock") than the current method using actual flight hours as the basis for retirement times. Of course, the benefits of usage monitoring are dependent on how the aircraft is operated. Based on the mission flown for this aircraft, which is flying work crews to offshore oil platforms, the flight hours charged against retirement times could be reduced by 50% or greater. Thus, the operator would gain a considerable payback in reduced maintenance costs due to extension of retirement intervals.

The FCR technique, which only modifies the helicopter maneuver spectrum relative to the manufacturer's baseline, was considered more practical and lower risk to implement compared to the FLS technique. However, the FLS technique could be refined to overcome shortcomings found.

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FOREWORD

This report presents the results of Phase 2 of Contract NAS3-25455 which includes the evaluation of usage monitoring techniques for retirement of rotorcraft life-limited dynamic components. This research was co-sponsored by the U.S. Army Propulsion Directorate, Aviation Research and Technology Activity and NASA Lewis Research Center in Cleveland, Ohio, and the Federal Aviation Administration (FAA) Technical Center, Atlantic City International Airport, New Jersey. The U.S. Army Contracting Officer's Technical Representative at NASA Lewis was Dr. Robert Handschuh and FAA Technical Cognizance was under the direction of Mr. Wayne Shade at the FAA Technical Center.

This study was conducted by Bell Helicopter Textron Inc. (BHTI) with support from PHI for the data collection. The BHTI project engineer was Mr. Jim Cronkhite, the lead Fatigue engineer was Mr. Bill Dickson with Mr. Rex Hayden conducting the FCR evaluation and Mr. Scott Bielefeld conducting the FLS evaluation. The support team at PHI included Messrs. Harold Summers, Donnie Doucet, Britt Hanks, and Raylund Romero at Lafayette, Louisiana, and the maintenance and pilot staff at PHI Morgan City, Louisiana base where the HUMS trial aircraft is operated.

1. INTRODUCTION

This feasibility study has been conducted for and under the cognizance of the Federal Aviation Agency (FAA), the U. S. Army, and NASA under Contract Number NAS3-25455. This study evaluated the effectiveness of two usage monitoring techniques for predicting fatigue damage to life-limited components of the Model 412 helicopter and compares the results to the manufacturer's component lives predicted by the safe-life methodology while using recorded data from an independent flight trial. Specifically, this study compares the manufacturer's retirement lives determined for several Model 412 components to lives predicted from the Flight Condition Recognition (FCR) and Flight Load Synthesis (FLS) methods. Should the lives determined from the FCR and FLS methods be greater than the manufacturer's baseline lives, the result will be longer time in service for the component and a reduced maintenance cost to the operator. Conversely, shorter lives would indicate a more severe mission and benefit the operator by reducing risk and increasing safety, Figure 1.

The helicopter usage data that was used in this study was obtained from an independent flight trial program of a Health and Usage Monitoring System (HUMS) installed on a Model 412SP helicopter, S/N 36007, N7128R, being operated in the Gulf of Mexico while performing an offshore support mission for the oil industry. The purpose of the flight trial program which began in November 1993 was to perform a comprehensive evaluation of the HUMS in an actual operating environment and generate the flight data used for evaluation of usage monitoring techniques.

The four major HUMS monitoring functions are listed in Figure 2. The functional areas incorporated in the flight trial HUMS were: Rotor Track and Balance, Engine monitoring, Drive System monitoring, and Usage monitoring of life limited components. The "U" in HUMS representing usage was not incorporated but was being evaluated off-line using PC-based software and the flight data from the trial program. The data required for the usage monitoring evaluation was recorded in time history format using an optical disk recorder, Figure 3. The data was retrieved weekly from the operator and then routed to the manufacturer for processing. Because many of the parameters required for usage were already a part of the Flight Data Recorder (FDR), the addition of sensors specifically to monitor usage was minimized.

In this report, the acquisition and processing of the usage data is described followed by a discussion of methodology used. The FCR and FLS methods are then discussed and the two methods are compared to each other along with the manufacturer's baseline (safe-life) for the offshore support mission. The economic impact of the methodologies is presented in terms of possible maintenance credits to the operator and the resulting impact on cost of operation. Finally, some conclusions are drawn based on this study together with recommendations for future work related to usage monitoring development.

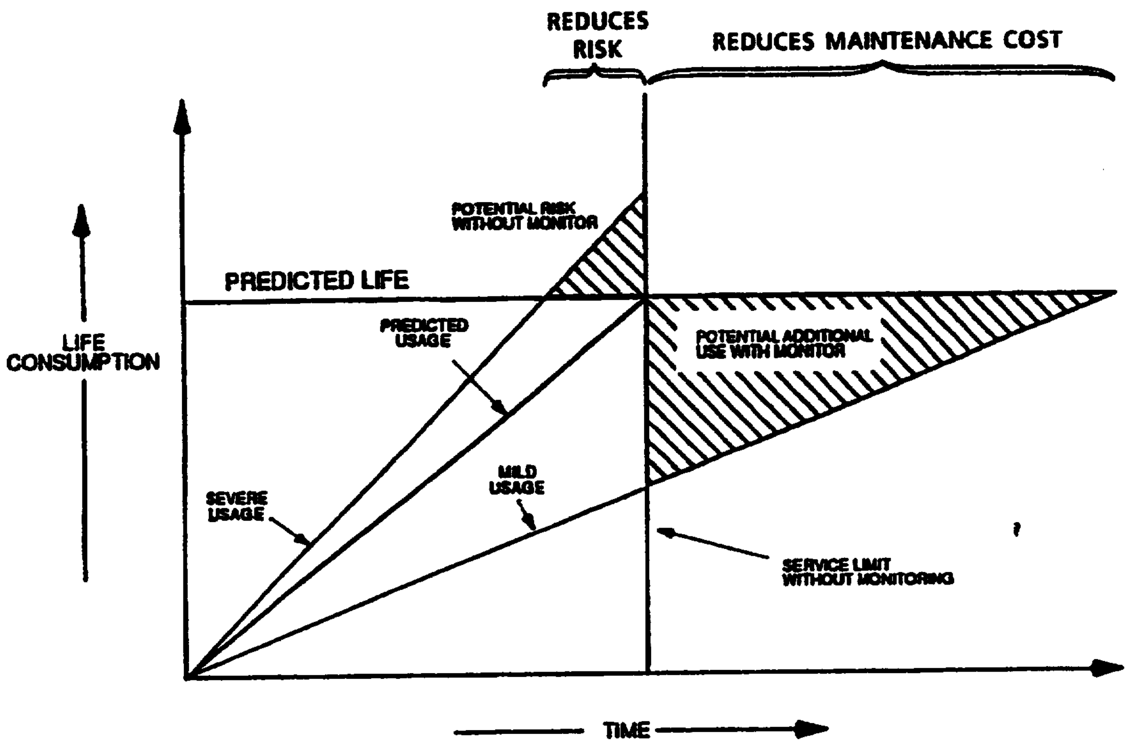

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Figure 1. Potential Benefits of Usage Monitoring

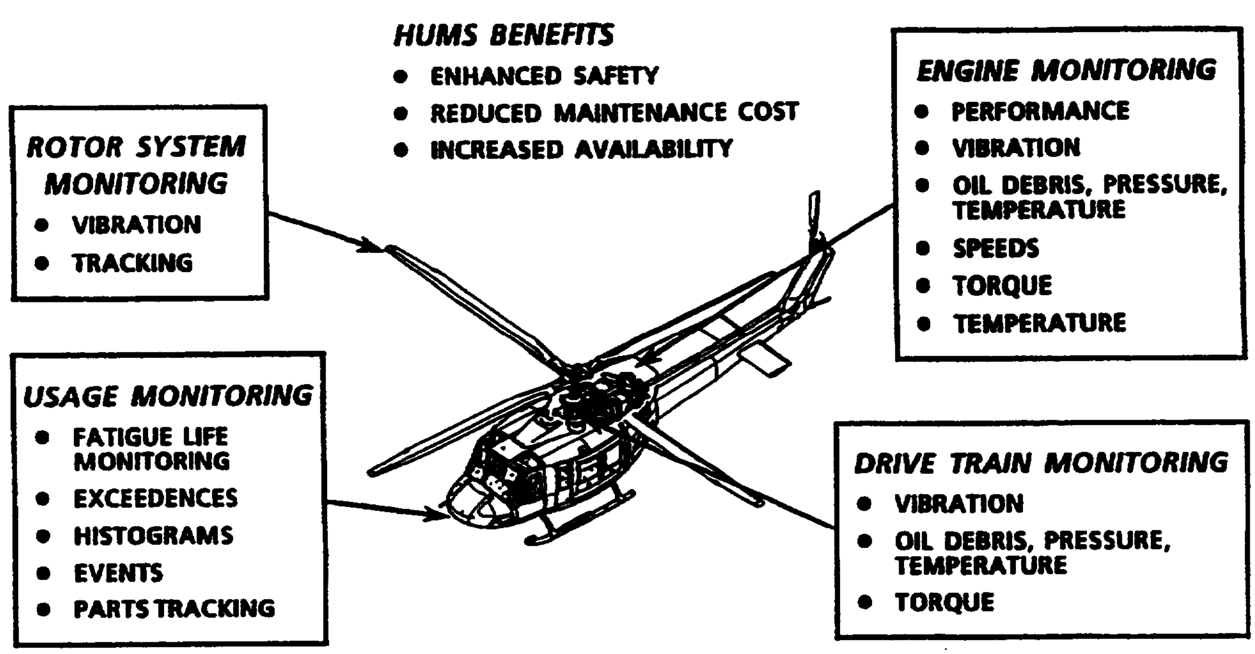

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Figure 2. HUMS Functions

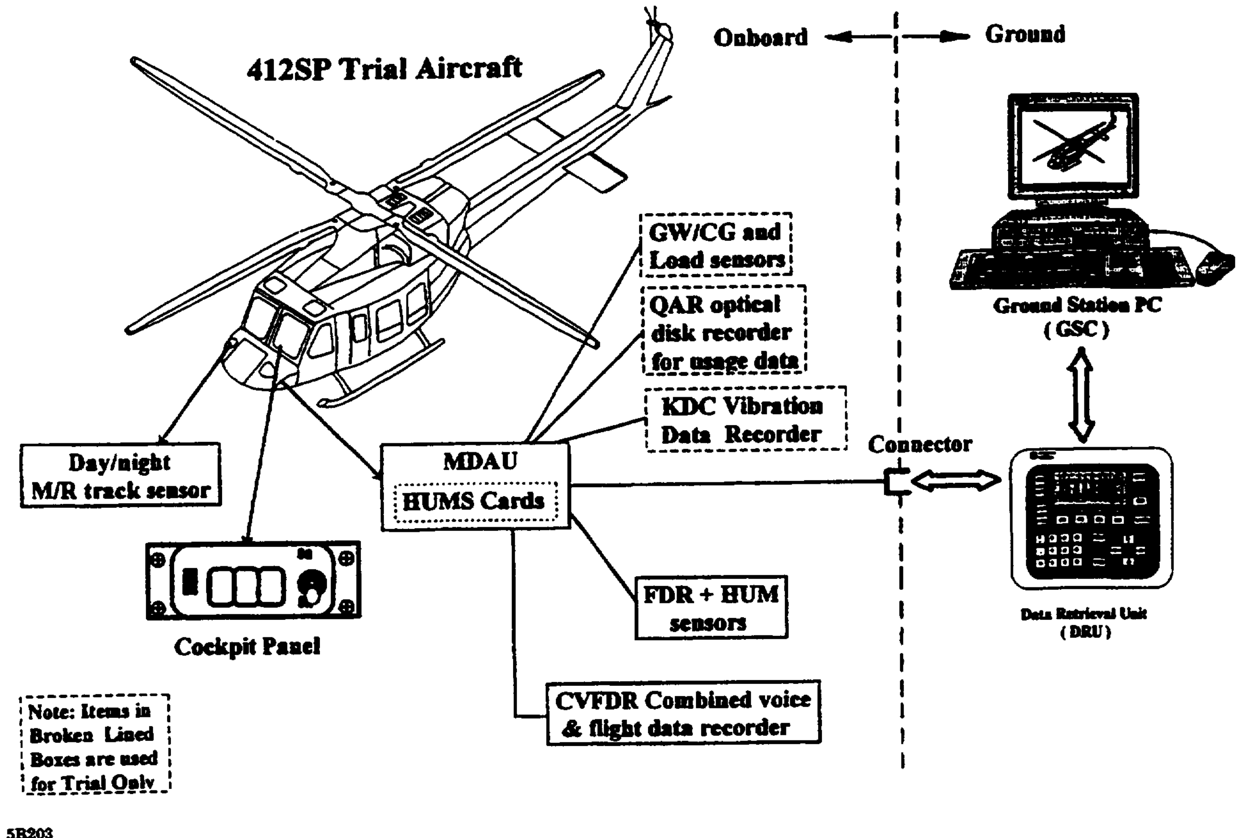


Figure 3. Integrated HUMS / FDR System

2. METHODOLOGY

2.1 USAGE METHODOLOGY

Two different approaches have been used in this study to determine component lives based on actual usage of the helicopter. One approach uses the Flight Condition Recognition (FCR) method while the other uses the Flight Load Synthesis (FLS) method. Both approaches use data from the onboard sensors as the input to predict component fatigue damage. During the flight trial the data was continuously recorded in time history format on an optical disk recorder for input to the ground-based PC data analysis system.

The components selected for the usage study are listed in Table 1 together with the current baseline or certification lives. Components manufactured from a variety of materials were selected for the main rotor, fixed controls, and rotating controls to show the sensitivity of the FCR and FLS methods to curve shapes used for the S-N curves. Table 2 is a list of both the helicopter parameters and load parameters recorded together with the applicable sample rate. Included in the monitored components added specifically for this study were three strain-gaged boost tubes in the fixed control system and a strain gaged L.H. forward fin spar member in the airframe. The fatigue damage in these components using the safe-life approach is compared to the predicted values for the FCR and FLS methods. The fin spar data was not suitable for inclusion because of a lack of correlation during the FLS effort. Four strain-gaged sensors added for the flight trial were used to measure helicopter gross weight and C.G. This information was used primarily for the FCR method, Figure 4.

The FCR method uses recorded data from the flight trial and derived algorithms to predict time in flight conditions performed by the operator during the flight trial. The FCR then accumulates the time for each condition. Subsequently, the damage rate associated with each flight condition (acquired from the manufacturer's certification database) is applied to the time accumulated in each flight condition to determine the accumulated damage. The algorithms used were derived from the helicopter manufacturer's certification load level data and checked using a scripted flight conducted early in the flight trial.

The FLS method uses a multiple linear regression approach to develop equation coefficients using selected parameters from the manufacturer's certification database. The "goodness" of the correlation is also predicted by comparing the measured data to the derived data. These same parameters recorded during the flight trial are used with the derived coefficients to predict oscillatory loads in both fixed and rotating helicopter components. Subsequently, the predicted loads are evaluated against the manufacturer determined fatigue strength (endurance limit) to determine the fatigue damage occurring for each monitored component.

Table 1. Components Selected for Usage Study

Description	Material	Life Based On	BHTI Recommended Life
Main Rotor Yoke	Titanium	Beamwise Bending at Yoke Station 4.8	5,000 Hours
Main Rotor Spindle	Steel	Axial Pitch Link Load	10,000 Hours
Rephasing Lever	Aluminum	Axial Pitch Link	5,000 Hours
Swashplate Inner Ring	Aluminum	R.H. Cyclic Boost Tube Load	10,000 Hours (1)
Collective Lever	Aluminum	Collective Boost Tube Load	10,000 Hours
Tail Fin Spar	Aluminum	Strain in Spar at Fin Station 69.0	N/A
Main Rotor Mast	Steel	Engine Power	10,000 Hours or 60,000 RIN (2)
Splined Plate Assembly	Steel	Engine Power	

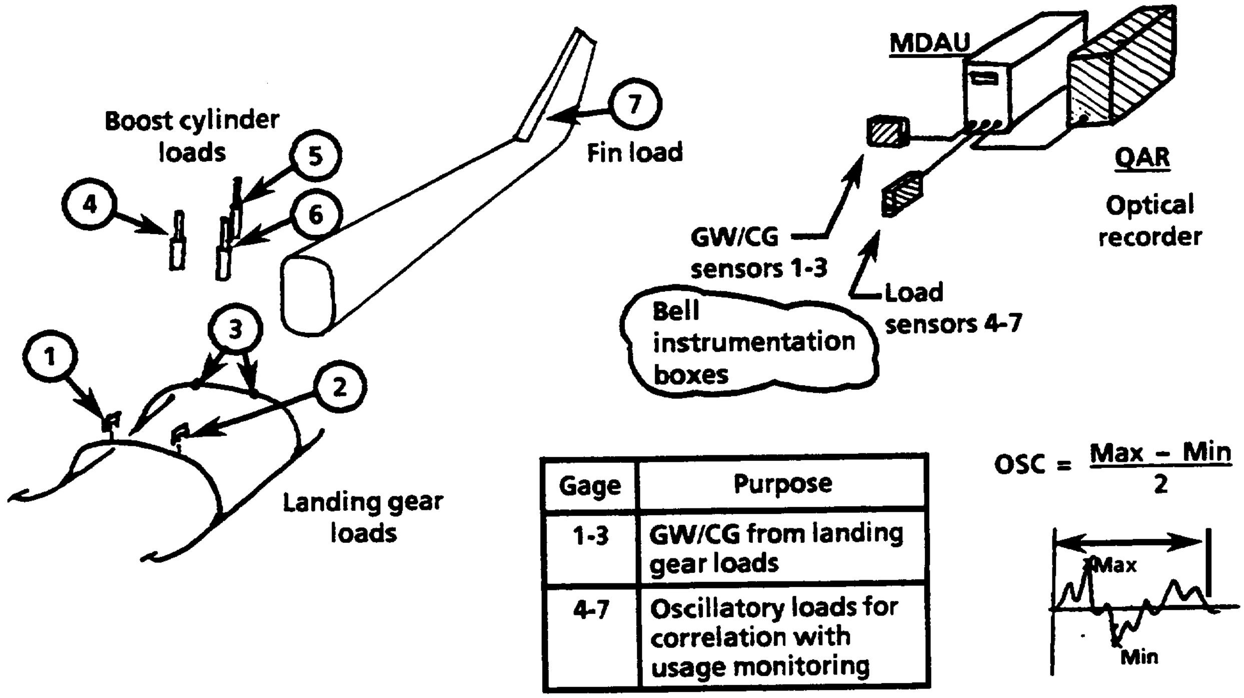
Notes

(1) Reduced from "unlimited" for purposes of this study.

(2) Determined from summed engine power using rainflow algorithm. Retirement Index Number (RIN) accumulates in service by manually counting each takeoff or lift event (Ref. ASB 412-94-81A).

Table 2. Model 412 HUMS Usage Parameters

Parameter	Sample Rate (Hz)
1. Calibrated Airspeed - CAS	1
2. Pressure Altitude - Hp	1
3. Outside Air Temperature - °C	1
4. Magnetic Heading - MH	1
5. Vertical C.G. Acceleration - N_Z	8
6. Pitch Attitude - θ	2
7. Roll Attitude - β	2
8. Altitude Rate - Climb (RC) or Descent (RD)	2
9. Main Rotor RPM - Nr	2
10. Engine Torque - T_1 or T_2	1
11. Collective Stick Position - COL	2
12. Longitudinal Cyclic Stick Position - F/A	2
13. Lateral Cyclic Stock Position - LAT	2
14. Pedal Position - PED	2
15. LH Cyclic Boost Load - LCL*	2
16. RH Cyclic Boost Load - RCL*	2
17. Collective Boost Load - CBL*	2
18. LH Forward Fin Spar Stress - LHF*	2
19. LH Forward Gross Weight Sensor*	1
20. RH Forward Gross Weight Sensor*	1
21. LH Aft Gross Weight Sensor*	1
22. RH Aft Gross Weight Sensor*	1
* Added for Usage Study	



Gage	Purpose
1-3	GW/CG from landing gear loads
4-7	Oscillatory loads for correlation with usage monitoring

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Figure 4. Additional Equipment and Sensors for Usage Data Collection

During the flight trial, four different methods of determining component fatigue damage were utilized:

1. Derived spectrum (FCR);
2. Derived loads (FLS);
3. Directly measured loads;
4. Hours logged by the operator.

Methods 1 and 2 are being evaluated and compared to the reference methods 3 and 4, Figure 5. It was necessary to accelerate the damage rates to better focus on the variation in component usage during the flight trial. This was accomplished by calculating damage with an adjusted endurance limit which resulted in a component life equal to that recommended in the original certification effort by the manufacturer. This same adjusted endurance limit was also used to calculate component damage for the FCR and FLS methods. This then permits a direct comparison of the values obtained from each method.

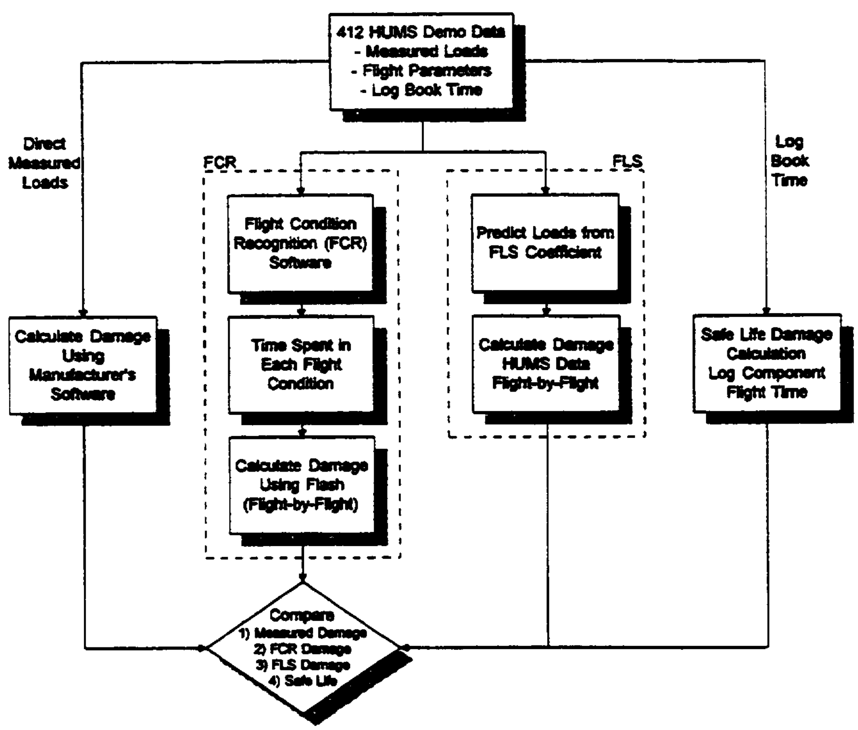

Figure 5. Usage Methodology Plan

2.2 FATIGUE METHODOLOGY

For both the FCR and FLS, the safe-life fatigue methodology has been used to determine component lives for this study. The lives of the selected components were determined using the three elements necessary to calculate a fatigue life. These elements are:

1. The fatigue strength of the component as determined by the manufacturer. Generally this takes the form of an S-N curve to define the strength distribution of the part on a load or stress vs cycles basis. A singular value, known as the endurance limit, is established which is the load or stress below which a component should not fail.

2. The component loads or stresses which occur during flight. The manufacturer conducts a comprehensive flight strain survey in which key dynamic components are strain gaged. The helicopter is flown through a selected list of flight conditions which constitute the expected operational flight spectrum. The loads occurring in the key instrumented components are recorded and stored for any future analysis connected with the helicopter certification effort.

3. The operational maneuvers which the helicopter will experience during the performance of its mission. This is commonly called the operational spectrum and per the FAA regulations should be conservative in nature i.e., more severe than any expected operation. Each maneuver is assigned a percentage of operating time and the total of all maneuver times should be 100%.

When all three of the above ingredients are known, a component life can be determined using Miner's Cumulative Damage Theory. Basically, this theory says that for any component, the fraction consisting of the cycles "n" allocated to a particular flight condition in the spectrum (usually a function of the rotor 1/P, 2/P, etc and percent time) divided by total load/stress cycles to failure "N" (determined from the S/N curve for each occurring oscillatory flight load) is the fatigue damage which has been incurred. The cumulative sum of these fractions for a given time is the damage which has occurred in the particular component in that time. This fraction is usually expressed as $\Sigma(\eta/N)$ over the number of flight conditions in the spectrum.

An overview of the procedures which have been employed in this study is provided below. Both the FCR and FLS method are described in more detail later in this report. Component lives determined using the flight trial recorded data and the FCR and FLS methods are also presented together with a comparison of the lives using the manufacturer's baseline certification loads. Distributions of the oscillatory loads occurring in the monitored components are also presented.

The FCR method simply replaces the spectrum which was assumed at the time of certification with the actual spectrum as determined for the mission being flown by the particular helicopter. In the case of FCR, the algorithms developed from the manufacturer's data allow determination of the time actually flown in each maneuver during the flight trial. The determination of the component damage then proceeds as described above with all other aspects of the process remaining unchanged. The S/N curve for the component is used as are the certification loads

from the manufacturer's database. This method does not decrease the conservatism built into the current fatigue life determination. FCR represents the least departure from methods currently being used by the manufacturer to determine component fatigue lives.

The FLS method is more of a departure from the current fatigue life determination process. Here, a mathematical relationship is established between values of certain easily measurable helicopter flight parameters, e.g., airspeed, altitude, load factor, stick positions, etc., and the key dynamic components of the helicopter which are not directly measured as easily. Components in the rotor or rotating system cannot be practically measured continuously because the information must be passed from the rotating to the fixed reference which requires an unreliable device such as a slip ring. The ability to predict these loads using parameters which are normally available on the helicopter in the fixed reference means that it would no longer be necessary to know the operational spectrum of the aircraft. The loads derived using FLS are simply used together with the S/N relationship to directly calculate component damage. The FLS method has a potentially higher probability of error than the FCR method in that the loads are mathematically derived instead of using the measured loads from the manufacturer's data base. However, the manufacturer's database of helicopter parameters and loads is used to derive the coefficients used in the correlation technique. This determination of the coefficients becomes the most critical part of the FLS method and must employ a certain degree of conservatism. FLS has an advantage in being able to identify flight loads for conditions which may not have been anticipated.

3. ACQUISITION AND PROCESSING OF FLIGHT TRIAL DATA

The HUMS trial was officially launched on November 26, 1993, from the helicopter operator's base as described in Reference 1. Table 2 presents the data parameters related to usage which were recorded continuously on optical disk onboard the trial helicopter between November 1993 and October 1994. Valid data for usage purposes was available from February 1994 through October 1994. Data used in this study totaled 583 hours consisting of data recorded in 18 weeks of flying from February to June of 1994. The data was recorded in optical format onboard the helicopter using a magnetic optical Quick Access Recorder (QAR). The disks were removed from the helicopter at about one week intervals and replaced with a blank disk. The disk containing the recorded data and written reports from the operator were forwarded to the manufacturer for processing and analysis.

Figure 6 presents a flowchart that details the data processing steps which were performed by the manufacturer. The first step in the data processing was a quick look at the flight trials data on the optical disk using a PC-based software program called FLIDRAS. This program allowed scanning and plotting the data in time history format with engineering units assigned. This program was used as a screening device early in the program to quickly determine any problems requiring immediate attention. As a result, several problems concerning the recorded data were diagnosed early and solved with little or no interruption in the program.

The second step in the data processing was to transfer the data on the optical disk to the manufacturer's mainframe VAX computer. The data was then processed and archived on the manufacturer's flight data file for subsequent analysis. The processing included conversion to correct engineering units and breaking the data into smaller more usable file sizes. The completed data was retained in the manufacturer's flight data file for input to the various PC-based analytical routines.

The manufacturer's flight data analysis system contained computer tools for plotting and listing data and facilitated reviewing and editing the data. Erroneous data was detected and eliminated using this software. This erroneous data only occurred when external source electrical power was applied to the helicopter and the rotors were not turning. A wildpoint edit routine was used to eliminate spurious data spikes for some parameters.

The end product of the manufacturer's data analysis routine were the input files for the PC-based FCR and FLS programs. A detailed description of the data editing and assembly is included in the discussion of the FCR and FLS methods later in this report.

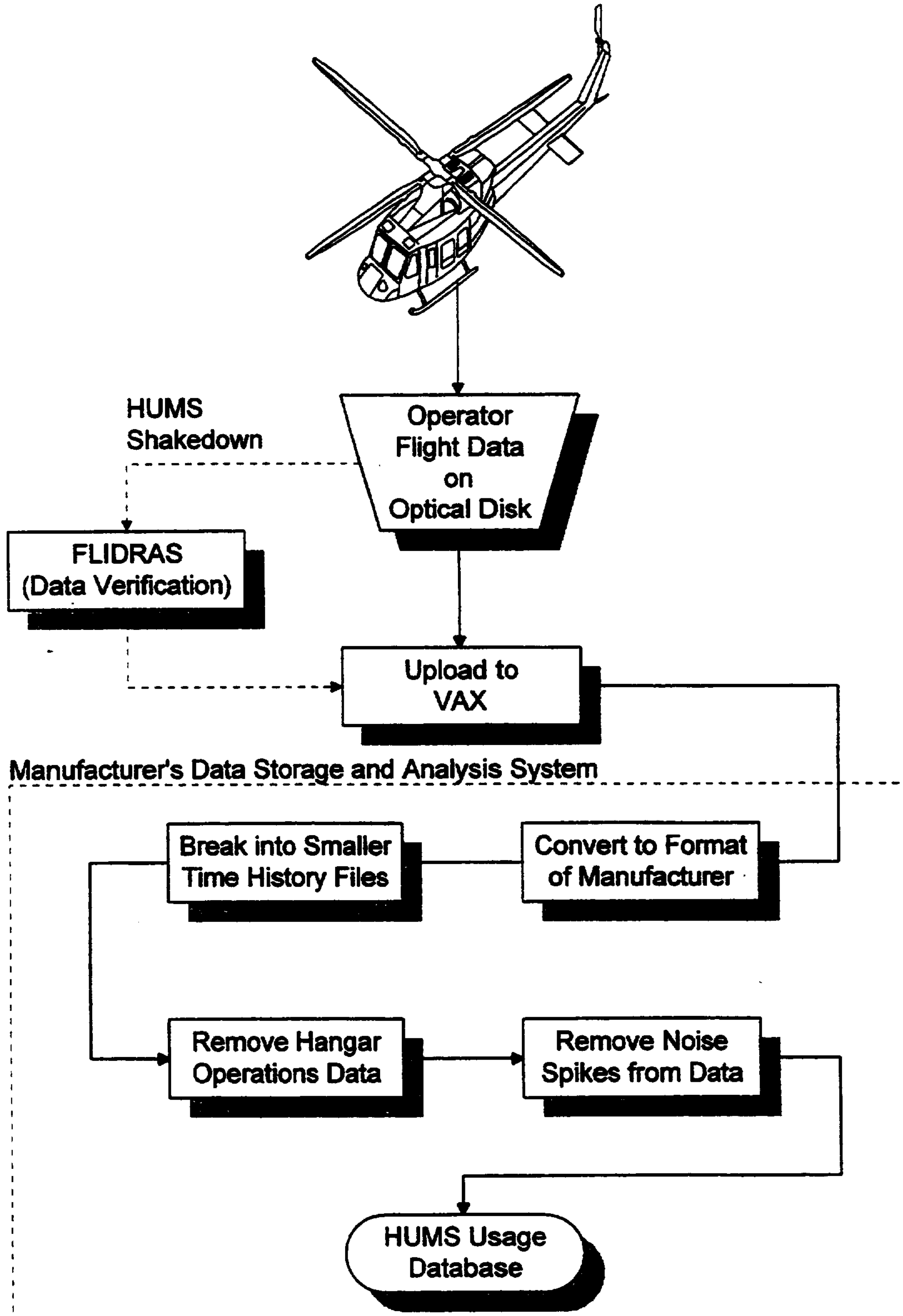

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Figure 6. Usage Data Processing

4. FLIGHT CONDITION RECOGNITION

4.1 FCR TECHNIQUE

Flight Condition Recognition (FCR) determines which flight condition from the contractor load level survey that the aircraft is performing at any given time. The output from the FCR program provides the actual operational spectrum of the aircraft. This actual operational spectrum replaces the assumed spectrum used in the manufactuer's safe-life calculations for recommended component retirement lives.

If an operator flies an aircraft less severely than the assumed spectrum used for determining recommended lives, then component lives for that helicopter will be increased. In other words, the hours accumulated for the components will be at a slower rate than the flight hours are accumulated, i.e., a "slow clock". This would result in increased savings for an operator.

If an operator flies a helicopter more severely than the assumed spectrum, then component lives will be decreased. The hours on the components would accumulate at a faster rate than the flight hours, i.e., a "fast clock". Components would be retired sooner but would result in greater safety for the operator.

4.2 FCR EVALUATION APPROACH

The FCR approach demonstrated during this study, as depicted in the flowchart of Figure 7, utilized the following items to determine time in each flight condition, and subsequently determine dynamic component usage:

1. Continuously recorded Basic Aircraft Parameter (BAP) data as presented in Table 3, items 1-19.

2. Deterministic computer program (ground based for this study) that checks basic aircraft parameter data against preprogrammed "normal ranges" to establish flight conditions. The output of this program is the cumulative time spent in each flight condition, divided into four gross weight ranges. This output can easily be converted into a spectrum. If a flight condition cannot be identified, that time is added to the unrecognized category.

3. Spectrum generated from FCR program. This spectrum was the result of analyzing 583 hours of operational data. Table 4 lists the conditions for which time was accumulated in the FCR program.

4. Manufacturer's fatigue life analysis computer program. The measured operational spectrum was used as the input into the analysis program to determine the actual damage rates for the components being evaluated. The actual life expended for each component was determined by

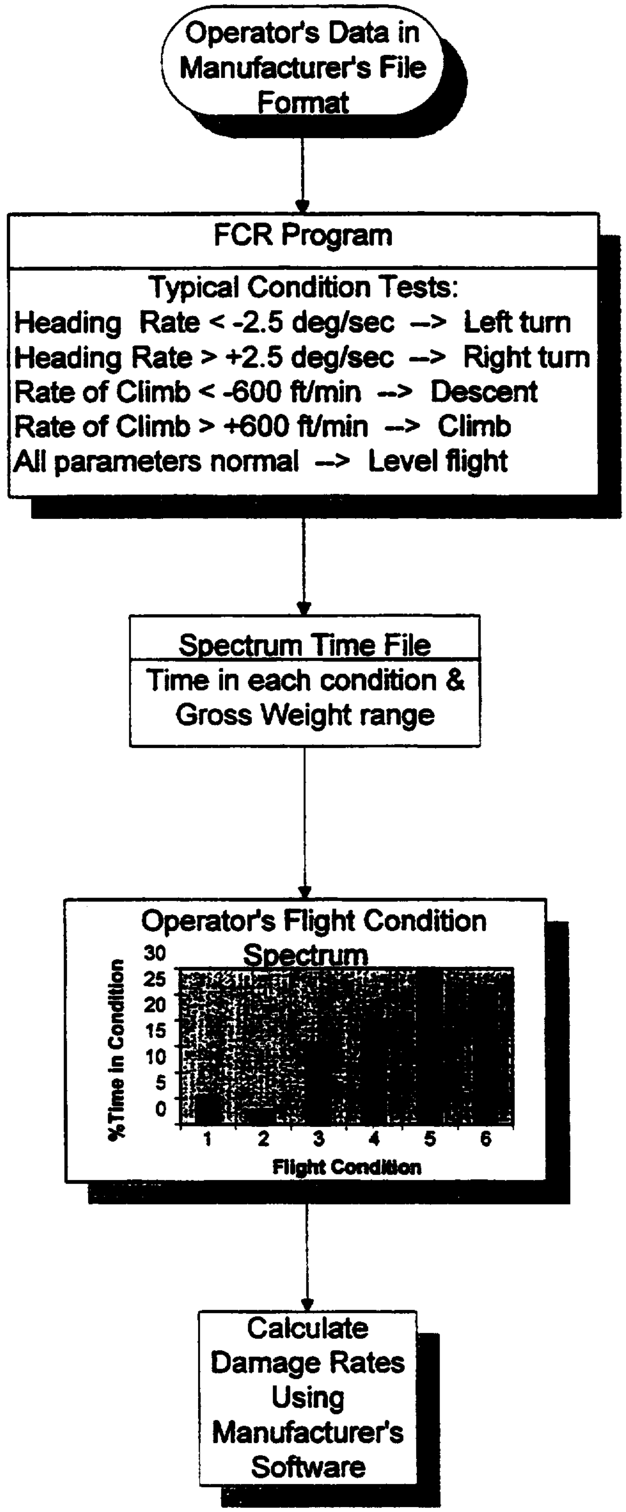

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Figure 7. Methodology for Flight Condition Recognition (FCR)

Table 3. Parameters Used for Flight Condition Recognition Program

Number	Parameter Name	Derived or Aircraft
1	Pitch	Aircraft Parameter
2	Roll	Aircraft Parameter
3	Vertical Velocity	Aircraft Parameter
4	RPM	Aircraft Parameter
5	Collective Stick Position	Aircraft Parameter
6	F/A Stick Position	Aircraft Parameter
7	Lateral Stick Position	Aircraft Parameter
8	Pedal Position	Aircraft Parameter
9	Normal Acceleration	Aircraft Parameter
10	Altitude	Aircraft Parameter
11	Left Forward GW	Aircraft Parameter
12	Left Aft GW	Aircraft Parameter
13	Right Forward GW	Aircraft Parameter
14	Right Aft GW	Aircraft Parameter
15	Airspeed	Aircraft Parameter
16	Left Engine Torque	Aircraft Parameter
17	Right Engine Torque	Aircraft Parameter
18	Heading	Aircraft Parameter
19	OAT	Aircraft Parameter
20	Heading Rate of Change	Derived Parameter
21	F/A Cyclic Rate of Change	Derived Parameter
22	Lateral Cyclic Rate of Change	Derived Parameter
23	Pedal Position Rate of Change	Derived Parameter
24	In-Air Flag	Derived Parameter
25	CG	Derived Parameter
26	Combined Engine Torque	Derived Parameter
27	Twin or Single Engine Flag	Derived Parameter
28	Airspeed Rate of Change	Derived Parameter
29	Elapsed Time, Seconds	Program Generated
30	Maneuver Number	Derived Parameter
31	Current Gross Weight	Derived Parameter
32	Moving Average, Vertical Velocity	Derived Parameter
33	V_H Fraction	Derived Parameter
34	Density Altitude	Derived Parameter

Table 4. Maneuvers Recognized by FCR Program

Number	Maneuver Name	Number	Maneuver Name
1	Rotor Start	28	High Speed Left Turn
2	On Ground	29	TE Partial Power Descent
3	Normal Takeoff	30	SE Partial Power Descent
4	Hover	31	TE - SE Transition Full Power Climb
5	Hover Right Turn	32	TE - SE Transition Level Flight
6	Hover Left Turn	33	SE - TE Partial Power Descent
7	Hover - Longitudinal Reversals	34	TE - Auto Transition in Low Speed
8	Hover - Lateral Reversals	35	TE - Auto Transition in High Speed
9	Hover - Pedal Reversals	36	Autorotation
10	Right Sideward Flight	37	TE Recovery From Auto
11	Left Sideward Flight	38	Autorotation Right Turn
12	Climbout (after takeoff)	39	Autorotation Left Turn
13	Twin Engine (TE) Landing	40	Vertical Ascent
14	Single Engine (SE) Landing	41	Vertical Descent
15	Level Flight, TE - 0.4 V_H	42	Low Speed Climbing Left Turn
16	Level Flight, TE - 0.6 V_H	43	High Speed Climbing Right Turn
17	Level Flight, TE - 0.8 V_H	44	High Speed Climbing Left Turn
18	Level Flight, TE - 0.9 V_H	45	Low Speed Descending Right Turn
19	Level Flight, TE - 1.0 V_H	46	Low Speed Descending Left Turn
20	Level Flight, TE - $>$ 1.0 V_H	47	High Speed Descending Right Turn
21	TE Full Power Climb	48	High Speed Descending Left Turn
22	SE Full Power Climb	49	Low Speed Climbing Right Turn
23	Low Speed Cyclic Pullup	50	Unrecognized
24	High Speed Cyclic Pullup	51	Shutdown
25	Low Speed Right Turn	52	High Speed Climb
26	Low Speed Left Turn	53	Dive Greater than 0.8 V_H Airspeed
27	High Speed Right Turn		

multiplying the flight hours by the appropriate damage rate. The unrecognized category time was multiplied by the damage rate for the most damaging maneuver in the operational spectrum. If the damage rate predicted would result in a component life greater than 25,000 hours being calculated, then a default rate which would result in retirement after 25,000 hours of flight time was used.

The FCR program was verified by comparing the results of a known flight maneuver sequence with the chronological log file output of the FCR program. The known flight maneuver sequence was obtained from a scripted flight conducted on March 12, 1994, using the trial helicopter. The requested flight sequence is presented in Table 5, and Table 6 presents a comparison of the requested flight sequence to the output from the FCR log file and includes all unrecognized time.

Within the FCR program, certain derived parameters were created. Table 3 presents a total list of input Basic Aircraft Parameter (BAP), Items 1-19, and derived parameters, Items 20-34, used for flight condition recognition. The parameters were selected so that the FCR program could be adapted to any helicopter by adjusting the values of the normal ranges. The internal process rate of the FCR program was two samples per second, and the program was designed to identify the 53 maneuvers listed in Table 4. Figure 8 presents a block diagram of the FCR program structure. The FCR computer program was used to process each week of data collected from the flight trial. Input data files were created using the manufacturer's flight data analysis software, and consisted of all aircraft parameters listed in Table 3. Output files consisted of:

1. A log file of all maneuvers performed, in chronological order
2. A spectrum time file of time in each maneuver and gross weight range
3. A time history data file with a user selected output
4. An operation's file that documented takeoff times, gross weight and C.G. at takeoff, and average flight times.

The normal ranges, and any other algorithms, were developed from actual maneuver data flown on the same model aircraft during the load level survey conducted by the manufacturer. Normal ranges were determined for the following parameters:

1. Magnetic heading rate of change
2. Rate of climb (descent), also known as vertical velocity
3. Pitch attitude
4. Roll attitude
5. Load factor

Table 5. Required Maneuvers for the Scripted Flight

Condition	Description
1	Rotor start to flight idle (Note clock time at start)
2	Stabilized idle (2 or 3 minutes)
3	Increase RPM to 100% (30 seconds to 1 minute)
4	Vertical takeoff to stabilized hover (Hold heading, 5-20 ft skid clearance)
5	Hover taxi in prep for takeoff
6	Takeoff and accel to climb airspeed (60-70 kt)
7	Stabilized climb to 800 to 1,000 ft above ground level
8	Level flight at 80 kt (2-4 minutes)
9	Accel to 115 kt
10	Level flight at 115 (2-4 minutes)
11	Right turn at 115 kt (35° to 45° bank angle, ~ 90° heading change while maintaining altitude)
12	Level flight at 115 kt
13	Left turn at 115 kt (Same characteristics as right turn)
14	Level flight at 115 kt
15	Climb an additional 500 to 1,000 ft
16	Level flight at 90 kt
17	Pushover to V_{NE} dive (140 kt) for 30 seconds duration
18	Cyclic pullout and decrease airspeed to 80 kt level flight
19	S-turns (right/left/right/level at 80 kt)
20	Heading changes/cruise as required to return to base
21	Descent for landing
22	Flare to stable hover
23	Right sideward flight
24	Stabilized hover
25	Left sideward flight
26	Stabilized hover
27	Hover taxi to landing spot
28	180° right hovering turn
29	Land/flight idle
30	Shutdown with collective (Note clock time at shutdown)

Table 6. Comparison of Scripted Flight and FCR Log Output

Clock Time	Elapsed Time	Scripted Flight Requirement	FCR Log Output	Time - In Seconds
14:51:08	0:00	Takeoff	Normal Takeoff	1
14:51:09	0:01	Hover Taxi in Prep for Takeoff	Hover	55
			Hover Turns	18
14:52:21	1:13	Stabilized Climb to 800 to 1000 ft above ground (approximately 75 sec)	TE Full Power Climb	63.5
			.6 V_H Level Flight	12
14:53:37	2:29	Level Flight, 80 kt (appx. 2 min)	.8 V_H Level Flight	107
			High Speed Left Turn	2
			Unrecognized	0.5
14:56:30	5:25	Level Flight, 115 kt (appx. 2 min)	.9 V_H Level Flight	77.5
			1.0 V_H Level Flight	31
			High Speed Climb	5
			High Speed Left Turn	13
			Unrecognized	1.5
14:58:42	7:34	Right Turn, 115 kt (appx 35 sec)	Moderate Right Turn	25.5
			High Speed Right Turn	4
			Unrecognized	0.5
14:59:12	8:04	Level Flight, 115 kt (appx 75 sec)	.9 V_H Level Flight	62.5
			1.0 V_H Level Flight	18.5
			High Speed Climb	1
15:00:34	9:26	Left Turn, 115 kt (appx 22 sec)	High Speed Left Turn	15
			Unrecognized	0.5
15:00:49	9:41	Level Flight, 115 kt (appx 40 sec)	.8 V_H Level Flight	1.5
			.9 V_H Level Flight	34
			High Speed Left Turn	4
			Unrecognized	1
15:01:30	10:22	No Requirements - Data verified pullup was performed	Low Speed Cyclic Pullup	24
15:01:54	10:47	Climb an Additional 500 - 1000 ft	TE Full Power Climb	3
			High Speed Climb	15
			Unrecognized	0.5
15:05:22	14:15	Pushover to V_{NE} Dive (appx 30 sec)	Dive Greater than .8 V_H	31
15:05:53	14:46	Cyclic Pullout	High Speed Cyclic Pullout	32
15:07:39	16:31	S-turns (right/left/right/level), 80 kt (appx. 108 sec)	Right Turn Maneuvers	32.5
			Level Flight	25
			Left Turn Maneuvers	48
15:20:29	29:21	Landing	Landing, TE	1

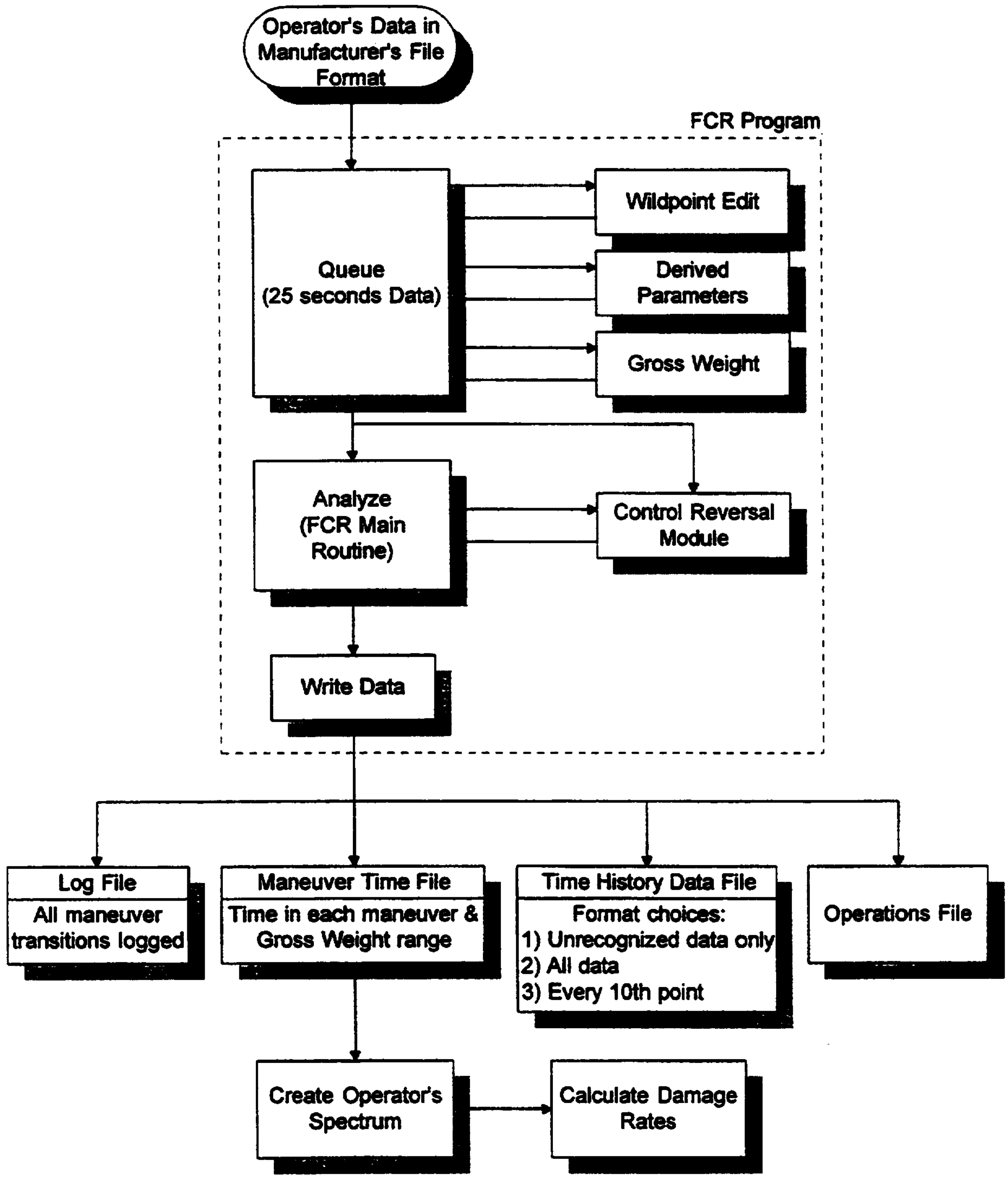

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Figure 8. Structure of Flight Condition Recognition (FCR) Program

6. Control position rates

7. Engine torques.

Maneuvers were defined when one or more parameters were out of the normal ranges.

The FCR program read data into a queue where 25 seconds (50 datapoints) were accumulated. Each cycle of the program read in one complete sample of data into the queue, and deleted the oldest sample from the queue. All input data was processed through a wildpoint edit module to remove spurious intermittent spikes recorded on the pitch and roll data. Data obtained from synchro channels were the only channels affected by the data spikes. Next, the derived parameter module processed the data. Most derived parameters were rate of change of a parent parameter, and were determined by simple time differentiation. The only special case was the heading parameter, which had a discontinuity at the 0 to 360 degree point. This was handled with a special set of instructions in the code. Another derived parameter was the moving average of vertical velocity. This parameter was used to smooth out the coarseness in the vertical velocity, so algorithm performance would be more stable.

Calculations were next performed by the gross weight module. The gross weight module determined:

1. Combined gross weight

2. When the aircraft took off (in-air flag)

3. C.G. at time of takeoff

4. Fuel burn adjustment

5. V_H, V_H fraction, and density altitude.

Combined gross weight was a function of the sum of all four gross weight sensors on the ground, minus a constant to account for the sum of their in-air values. The on-ground sum was taken as an average for the time period of 25 seconds to 14 seconds before takeoff. This average sum was considered valid if the rotor RPM was greater than 97%, and the collective setting was less than 5%. If these conditions were not met, an algorithm was used to correct the raw gross weight value for the collective setting. The sum of the in air gross weight values was a constant 1500 $\pm$100 lb. The "In-Air" flag was determined by the total sum of the gross weight sensors. When the total sum of the sensors was less than 3700 lb, the helicopter was considered to be in the air. Helicopter C.G. was calculated for the same time period as the combined gross weight, using a simple sum of moments equation. The fuel burn equation was a function of pressure altitude, outside air temperature, and combined engine torque.

The derived parameter V_H fraction calculated the ratio of the current value of calibrated airspeed to the allowable V_H airspeed. The V_H airspeed equation calculated a maximum airspeed based on density altitude and gross weight.

4.3 FCR RESULTS

The spectrum time files created for each week of processed data by the FCR program were merged together to create a cumulative operational spectrum, which is presented in Table 7. Table 8 presents a comparison of the time at each condition for the original certification spectrum and the derived operator spectrum. Breakdowns for gross weight, RPM, ground time, and flight time are presented in Table 9.

The damage rates were determined by using the cumulative spectrum as input into the manufacturer's analysis program. Fatigue life expended for the components being evaluated were determined using these damage rates, and a summary is presented in Table 10. Figures 9 through 13 present plots that compare component fatigue life used based on logged flight hours to fatigue life used based on FCR methodology.

Table 7. Cumulative Maneuver Spectrum for Model 412 Trial Helicopter

Maneuver	On Ground	Gross Weight Ranges (lb): Less than 8,000	8,000 to 10,000	10,000 to 12,500	Greater than 12,500	Totals: Hours	Pcts	Time Applied to Certification Spectrum Condition Number
		Time, in Hours						
Rotor Start	0.1672	0.0000	0.0000	0.0000	0.0000			
On Ground	132.6290	0.0000	0.0000	0.0000	0.0000			
Normal Takeoff	0.0000	0.0131	0.0778	0.0968	0.0029	0.1906	0.0423	14
Hover	0.0000	2.9811	4.0557	5.2481	0.1015	12.3864	2.7504	4, 5
Hover Right Turn	0.0000	0.0650	0.8782	0.9803	0.0267	1.9501	0.4330	6
Hover Left Turn	0.0000	0.0667	0.8436	0.7836	0.0213	1.7151	0.3809	7
Hover - Longitudinal Reversals	0.0000	0.0540	0.0576	0.0362	0.0011	0.1490	0.0331	8
Hover - Lateral Reversals	0.0000	0.0636	0.0475	00504	0.0000	0.1615	0.0359	9
Hover - Pedal Reversals	0.0000	0.1146	0.1883	0.1281	0.0047	0.4357	0.0967	10
Right Sideward Flight	0.0000	0.0028	0.0646	0.1032	0.0003	0.1708	0.0379	11
Left Sideward Flight	0.0000	0.0225	0.2378	0.1753	0.0042	0.4397	0.0976	12
Landing TE	0.0000	0.1383	0.0107	0.0029	0.0008	0.1528	0.0339	15
Landing SE	0.0000	0.0254	0.0094	0.0029	0.0001	0.0379	0.0084	16
Level Flight, TE - 0.4 V_H	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	17, 18
Level Flight, TE - 0.6 V_H	0.0000	0.2013	3.7442	5.8276	0.0865	9.8596	2.1893	19, 20
Level Flight, TE - 0.8 V_H	0.0000	0.2590	4.9946	9.7581	0.1563	15.1679	3.3681	21, 22
Level Flight, TE - 0.9 V_H	0.0000	0.7246	11.4329	37.7179	0.3308	50.2063	11.1484	23, 24
Level Flight, TE - 1.0 V_H	0.0000	2.6329	70.5154	209.5056	1.9885	284.6424	63.2055	25, 26
Level Flight, TE - > 1.0 V_H	0.0000	0.0192	3.3325	6.5903	0.2167	10.1586	2.2557	27, 28
TE Full Power Climb	0.0000	0.0917	2.3408	4.5738	0.0774	7.0836	1.5729	29
SE Full Power Climb	0.0000	0.0000	0.0036	0.0024	0.0000	0.0060	0.0013	30

Table 7. Cumulative Maneuver Spectrum for Model 412 Trial Helicopter (Continued)

Maneuver	On Ground	Gross Weight Ranges (lb)				Totals		Time Applied to Certification Spectrum Condition Number
		Less than 8,000	8,000 to 10,000	10,000 to 12,500	Greater than 12,500	Hours	Pcts	
		Time, in Hours						
Low Speed Cyclic Pullup	0.0000	0.0119	0.2628	0.1133	0.0000	0.3881	0.0862	31
High Speed Cyclic Pullup	0.0000	0.0006	0.0556	0.0260	0.0000	0.0821	0.0182	32
Low Speed Right Turn	0.0000	0.0600	1.9126	2.9092	0.0368	4.9186	1.0922	34
Low Speed Descending Right Turn	0.0000	0.0008	0.0722	0.0549	0.0000	0.1279	0.0284	34
Low Speed Climbing Right Turn	0.0000	0.0114	0.2508	0.2736	0.0117	0.5475	0.1216	34
High Speed Right Turn	0.0000	0.0139	0.4253	0.5422	0.0008	0.9822	0.2181	35
High Speed Descending Right Turn	0.0000	0.0003	0.0117	0.0414	0.0097	0.0631	0.0140	35
High Speed Climbing Right Turn	0.0000	0.0021	0.0887	0.0914	0.0000	0.1822	0.0405	35
Low Speed Left Turn	0.0000	0.0335	0.4850	0.6674	0.0089	1.1947	0.2653	36
Low Speed Descending Left Turn	0.0000	0.0000	0.0846	0.1232	0.0047	0.2125	0.0472	36
Low Speed Climbing Left Turn	0.0000	0.0207	0.3787	0.3975	0.0000	0.7969	0.1770	36
High Speed Left Turn	0.0000	0.0461	0.6636	0.7444	0.0072	1.4614	0.3245	37
High Speed Descending Left Turn	0.0000	0.0053	0.0761	0.0488	0.0017	0.1318	0.0293	37
High Speed Climbing Left Turn	0.0000	0.0042	0.0896	0.0961	0.0011	0.1910	0.0424	37
TE Partial Power Descent	0.0000	0.0678	1.3358	1.3204	0.0275	2.7515	0.6110	42
SE Partial Power Descent	0.0000	0.0068	0.0957	0.0428	0.0000	0.1453	0.0323	43

Table 7. Cumulative Maneuver Spectrum for Model 412 Trial Helicopter (Concluded)

Maneuver	On Ground	Gross Weight Ranges (lb) Less than 8,000	8,000 to 10,000	10,000 to 12,500	Greater than 12,500	Totals Hours	Pcts	Time Applied to Certification Spectrum Condition Number
		Time, in Hours						
TE - SE Transition Full Power Climb	0.0000	0.0001	0.0006	0.0006	0.0000	0.0013	0.0003	44
TE - SE Transition Level Flight	0.0000	0.0006	0.0138	0.0147	0.0004	0.0294	0.0065	45
SE - TE Partial Power Descent	0.0000	0.0007	0.0135	0.0086	0.0000	0.0228	0.0051	46
TE - Auto Transition in Low Speed	0.0000	0.0000	0.0007	0.0004	0.0000	0.0011	0.0002	47
TE - Auto Transition in High Speed	0.0000	0.0000	0.0000	0.0003	0.0000	0.0003	0.0001	48
Autorotation	0.0000	0.0171	0.4489	0.3499	0.0006	0.8164	0.1813	42
TE Recovery from Auto	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	49
Autorotation Right Turn	0.0000	0.0000	0.0283	0.0293	0.0000	00.0576	0.0128	52
Autorotation Left Turn	0.0000	0.0000	0.0157	0.0164	0.0000	0.0321	0.0071	53
Vertical Ascent	0.0000	0.0222	0.1321	0.2400	0.0111	0.4054	0.0900	14
Vertical Descent	0.0000	0.0125	0.0518	0.1239	0.0112	0.1994	0.0443	15
Unrecognized	0.0000	0.0206	0.6208	0.6429	0.0196	1.3039	0.2895	(Note 1)
Shutdown	2.1074	0.0000	0.0000	0.0000	0.0000	2.1074	0.4679	15
High Speed Climb	0.0000	0.0894	4.1590	16.7500	0.3571	21.3556	4.7421	29
Dive Greater than 0.8 V_H Airspeed	0.0000	0.1139	3.9543	10.6696	0.1829	14.9207	3.3132	42
TOTALS	134.9036	8.0381	118.5636	317.9224	3.7128	450.3442	100.0000	

(1) Time allocated to condition with largest damage rate.

Table 8. Comparison of Assumed Certification Spectrum and Derived Operator Spectrum

Flight Condition	Spectrum Comparison		
	Certification	Operator	Cond. No.
I. Ground Conditions			
A. Rotor Start	0.0000	0.0000	1
B. Ground Time (RPM 250 - 324)	0.0000	0.0000	2
C. Normal Shutdown W/Coll	0.0000	0.0000	3
II. IGE Maneuvers			
A. Hovering			
1. Steady @ 314 RPM	1.3000	0.5501	4
2. Steady @ 324 RPM	2.5950	2.2003	5
3. 90° Right Turn	0.0900	0.4330	6
4. 90° Left Turn	0.0900	0.3809	7
5. Control Reversal			
a. Longitudinal	0.0120	0.0331	8
b. Lateral	0.0120	0.0359	9
c. Rudder	0.0120	0.0968	10
B. Sideward Flight			
1. Right	0.3250	0.0379	11
2. Left	0.3250	0.0976	12
C. Rearward Flight	0.1300	0.0000	13
D. Norm T/O and Accel to Climb A/S	1.7510	0.1323	14
E. Norm Approach and Land			
1. Twin Engine	2.0450	0.5461	15
2. Single Engine	0.0430	0.0084	16
III. Forward Level Flight			
A. 0.4 V_H 314 RPM	0.8000	0.0000	17
324 RPM	0.2000	0.0000	18

Table 8. Comparison of Assumed Certification Spectrum and Derived Operator Spectrum (Continued)

Flight Condition	Spectrum Comparison		
	Certification	Operator	Cond. No.
B. 0.6 V_H 314 RPM	2.4000	0.4379	19
324 RPM	0.6000	1.7514	20
C. 0.8 V_H 314 RPM	12.0000	0.6736	21
324 RPM	3.0000	2.6945	22
D. 0.9 V_H 314 RPM	16.0000	2.2297	23
324 RPM	4.0000	8.9187	24
E. 1.0 V_H 314 RPM	30.4000	12.6411	25
324 RPM	7.6000	50.5644	26
F. V_{NE} 314 RPM	0.8000	0.4511	27
324 RPM	0.2000	1.8046	28
IV. Power-On Maneuvers			
A. Full Power Climbs			
1. Twin Engine	4.7500	6.3150	29
2. Single Engine	0.1200	0.0013	30
B. Cyclic Pullups			
1. 0.6 V_H	0.1500	0.0862	31
2. 0.9 V_H	0.0500	0.0182	32
C. Norm Accel from Climb A/S to 0.9 V_H	1.0000	0.0000	33
D. Turns			
1. Right			
a. 0.6 V_H	1.0000	1.2422	34
b. 0.9 V_H	1.0000	0.2726	35
2. Left			
a. 0.6 V_H	1.0000	0.4894	36
b. 0.9 V_H	1.0000	0.3962	37

Table 8. Comparison of Assumed Certification Spectrum and Derived Operator Spectrum (Concluded)

Flight Condition	Spectrum Comparison		
	Certification	Operator	Cond. No.
E. Cont Rev @ 0.9 V_H			
1. Longitudinal	0.0500	0.0000	38
2. Lateral	0.0500	0.0000	39
3. Rudder	0.0500	0.0000	40
F. Decel from 0.9 V_H to Descent A/S	0.1800	0.0000	41
G. Part Power Descent			
1. Twin Engine	2.6440	4.1055	42
2. Single Engine	0.1300	0.0323	43
V. Power Transitions			
A. Twin to Single Engine in Full Power Climb	0.0100	0.0003	44
B. Twin to Single Engine at 0.9 V_H	0.0100	0.0065	45
C. Single to Twin Engine in Power Descent	0.0100	0.0051	46
D. Twin Power to Auto			
1. 0.6 V_H	0.0050	0.0003	47
2. 0.9 V_H	0.0050	0.0001	48
E. Stab Auto to Twin Engine - Norm Auto A/S	0.0100	0.0000	49
VI. Autorotation Flight at V_{NE} (AR)			
A. Stab Forward Flight			
1. At Min RPM	0.0200	0.0000	50
2. At Max RPM	0.0200	0.0000	51
B. Turns			
1. Right	0.0030	0.0128	52
2. Left	0.0030	0.0071	53
VII. Unrecognized	0.0000	0.2895	
TOTAL	100.0000	100.0000	

Table 9. General Information

Category	Ranges	Percent Time
Flight Time vs. Ground Time	On Ground	22.8
	In Air	77.2
Gross Weight Breakdown	Less than 8,000 lb	1.8
	8,000 lb to 10,000 lb	26.4
	10,000 lb to 12,500 lb	70.8
	Greater than 12,500 lb	1
RPM	Less than 319 RPM	80
	Greater than 319 RPM	20
Density Altitude	Less than 3,000 ft	61
	3,000 To 6,000 ft	28
	Greater than 6,000 ft	11
General	Maneuvers	15
	Hover	2.8
	Level Flight Less than 0.9 VH	5.6
	Level Flight Greater than 0.9 VH	76.6

Table 10. Comparison of Fatigue Lives

Part Name	Life Usage in Hours			Life Usage Improvement Using FCR, %
	Logged Hr	FCR	Measured	
Collective Lever Assembly	450	228	135	97%
Swashplate Inner Ring Assembly	450	245	181	84%
Rephase Lever Assembly	450	116	N/A	288%
Main Rotor Spindle	450	180	N/A	150%
Main Rotor Yoke Assembly	450	280	N/A	61%

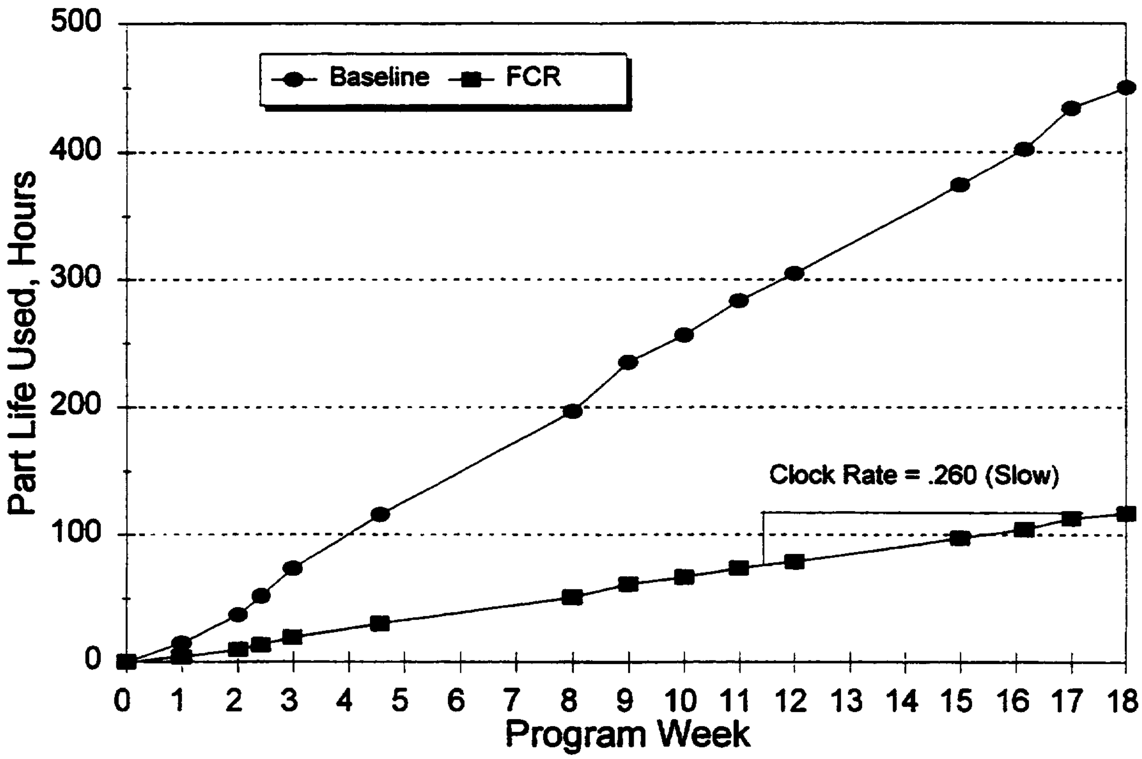

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Figure 9. Life Usage Comparison - Main Rotor Yoke

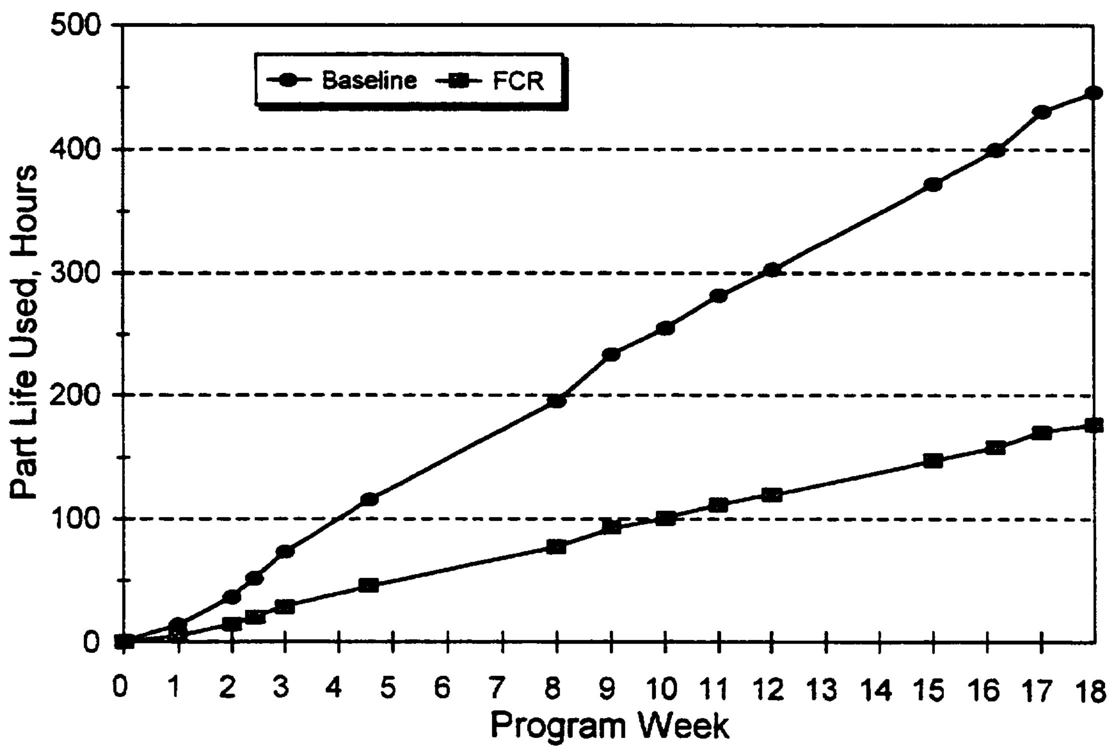

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Figure 10. Life Usage Comparison - Main Rotor Spindle

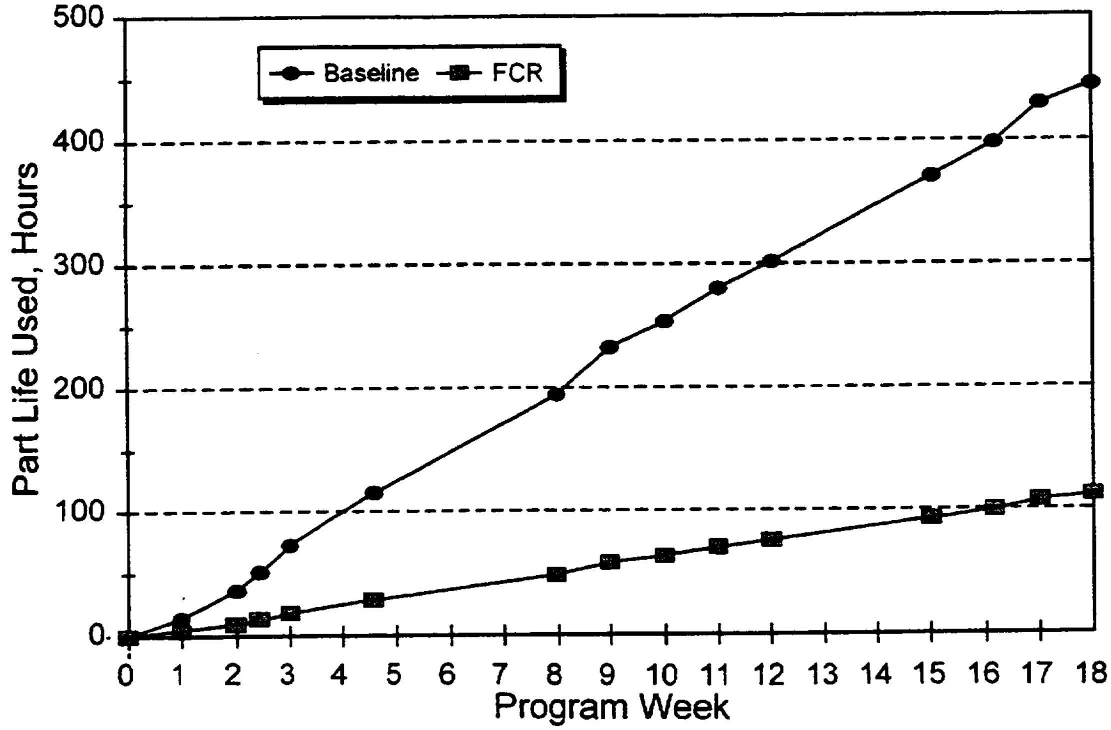

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Figure 11. Life Usage Comparison - Rephase Lever Assembly

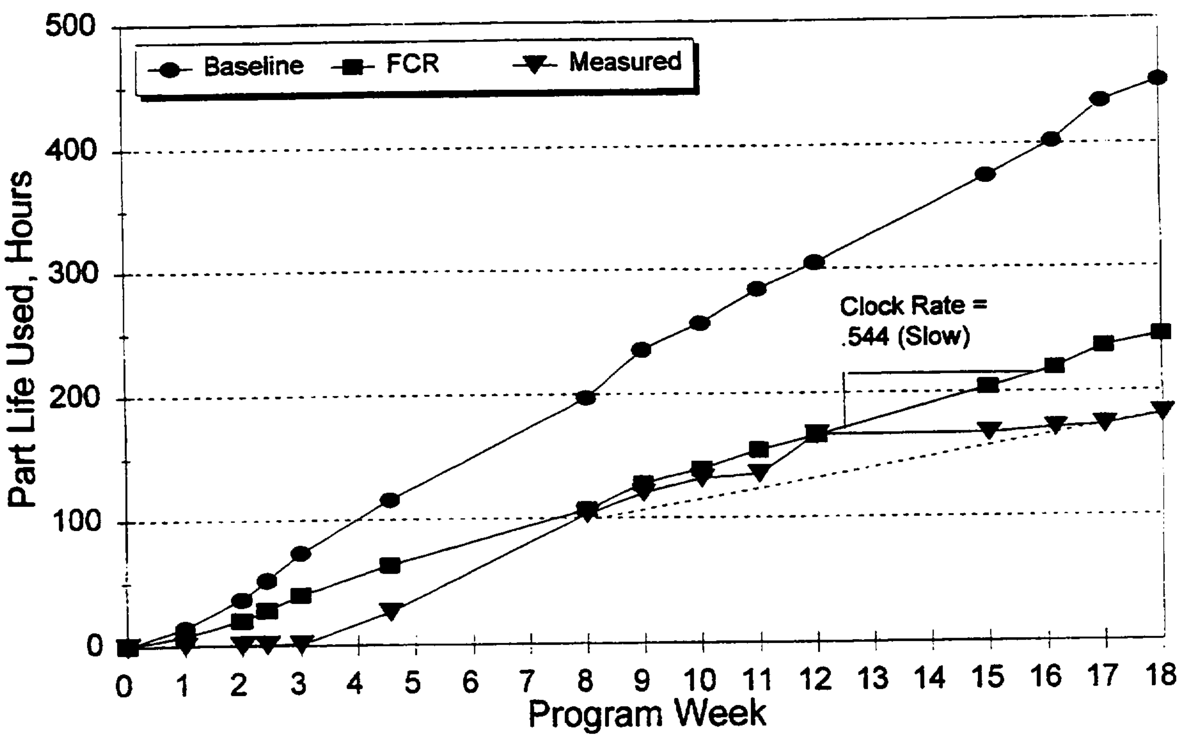

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Figure 12. Life Usage Comparison - Swashplate Inner Ring

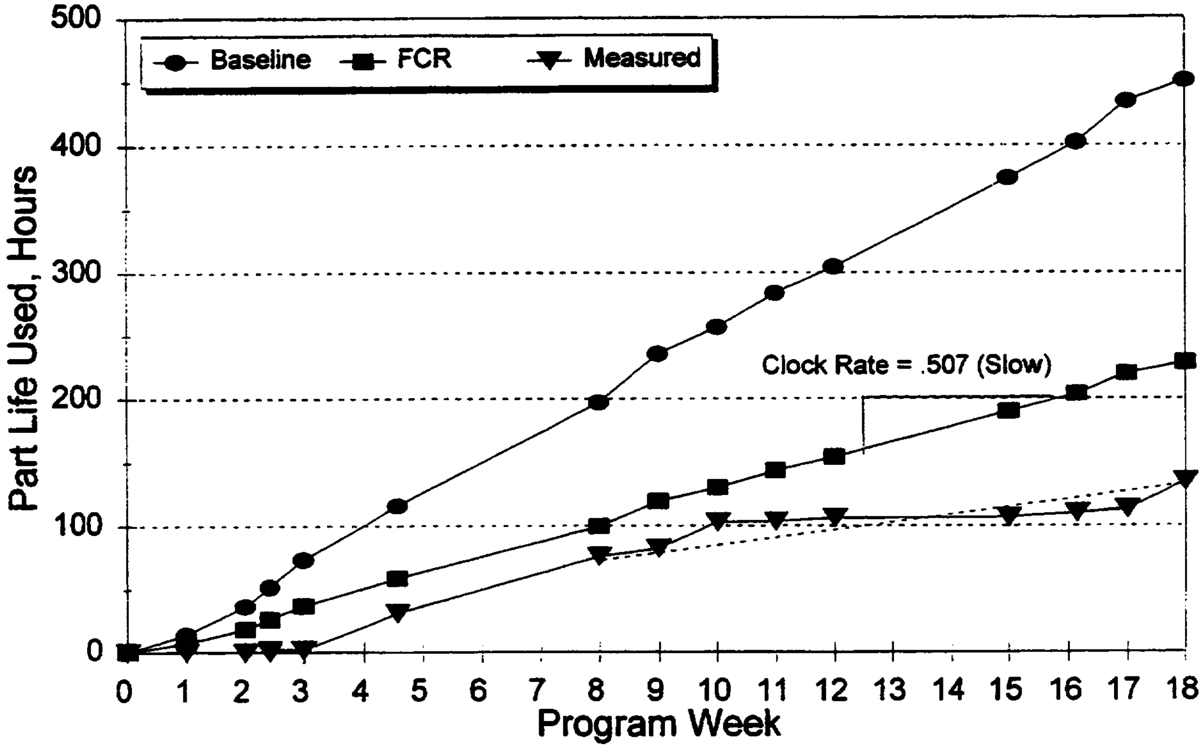

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Figure 13. Life Usage Comparison - Collective Lever

5. FLIGHT LOADS SYNTHESIS

5.1 FLS TECHNIQUE

Flight Loads Synthesis (FLS) attempts to predict the loads on non-instrumented fatigue critical components. A relationship is sought between the critical component loads and standard flight parameters such as airspeed, attitude, stick position, etc. Once this association is developed, critical component life can be defined as a function of common, quasi-static flight parameters.

The objective of FLS is the same as that for Flight Condition Recognition (FCR):

1. Lower the damage rates of those helicopters flying a more benign spectrum than that used by the airframe manufacturer to determine Safe-Life.

2. Recognize those cases where actual damage is occurring more quickly than that predicted by Safe-Life.

3. Guarantee conservatism and safety in the process.

At the foundation of fatigue life calculations is the S-N curve, which determines the number of cycles a component can tolerate for a given stress level. Ideally, the true load-cycle history would be recorded for each fatigue critical component and applied directly to the applicable S-N curve to calculate damage. However, this is not practical. Most fatigue critical components are in the rotating system and the means to transfer data to the fixed system are difficult and expensive to install and maintain. However, events in the fixed system do have a direct impact on loads in the rotating system. Current studies in FLS are investigating different methods of obtaining rotating system loads as a function of fixed system inputs.

FLS in this study focused on the development of a relationship between the oscillatory loads of fatigue critical components and common quasi-static flight parameters. It is important to note that the time-history data from the critical components were reduced to discrete oscillatory values, one data point for each rotor cycle. This approach differs from most FLS studies because the synthesis method does not have to relate the flight parameters to oscillatory loads as a function of time. Instead, only the maximum oscillatory load per rotor revolution is compared with the flight parameters. This removes all of the intricate phase relationships and makes FLS inherently easier while retaining the essential information required for damage calculations.

The three primary techniques that have been used in prior and ongoing FLS studies are holometrics, neural networks, and multiple linear regression (MLR), as described in References 2 through 5. The holometric method was dismissed for this project

since its output is a time-history trace, which is not a requirement for the damage calculation method used in this study. Neural network approaches, while showing promise, lack the maturity and tools that are currently available for MLR. In this study, equations were developed through multiple linear regression to calculate loads on six critical helicopter components.

Multiple linear regression attempts to represent a desired variable as a function of many other variables. The linear regression equations have the form.

$$y = c_0 + c_1f_1(x_1,x_2,...,x_n) + c_2f_2(x_1,x_2,...,x_n) + ... + c_mf_m(x_1,x_2,...,x_n)$$

The regression equation is linear with respect to the functions of x, but the functions themselves do not have to be linear. Ideally, all the applicable functions and their characteristics are known from a theoretical basis. For example, lift on a blade is known to be a function of velocity squared, angle of attack, etc. However, in cases such as HUMS, the relationship between each input variable and the output is not known, and a statistical model must be used. In a statistical model, many different regression equations are tried, and the tightness of the fit between the results of the equation and the actual recorded values are compared. The best regression is that which gives the strongest fit for a given number of terms in the equation. Care must be taken not to include variables that cannot logically influence the output. For example, engine temperature should have no bearing on rotor loads and is therefore not included.

The basic parameters used in the MLR procedure are given in Table 11.

5.2 FLS EVALUATION APPROACH

Multiple linear regression was performed using SAS®, a popular statistical analysis package that can handle regression procedures of the magnitude found in this project. The 14 helicopter parameters shown in Table 11 were used as inputs. In addition to these base variables, cross-products and squares of the parameters were generated. This gave SAS over 100 variable combinations from which to choose. During the generation of each regression, a multitude of different equations were compared for the tightness of fit between their output and the actual recorded value on the load-level survey helicopter. A sample equation is:

$$\text{pitch link} = c_0 + c_1(\text{pitch attitude}) + c_2(\text{roll rate})(\text{pedal position}) + ... + c_m(\text{rotor mast torque})^2$$

The R^2 value, which represents the fit of the regression, was used to compare the different equations. The best 30-term equations were selected for each component. Although these long equations have no physical justification, it was verified that each term in the regression equations was statistically significant. Over 6,000 points of data obtained in the load level survey were used in the regression development, so there was no problem with overfitting.

Table 11. Basic Parameters Used in Multiple Linear Regression (MLR) Procedure

Parameter Source in Manufacturer Certification Data	Description	Parameter Source on HUMS Demonstrator Helicopter	Units
30BB01	Yoke Beam Bending	equation	in-lb
30FA41	Pitch Link	equation	lb
10FA54	Collective Boost Tube	equation	lb
02SAL3	Fin Spar Strain, Station 69.0	equation	u-strain
10FA55	Left Boost Tube	equation	lb
10FA57	Right Boost Tube	equation	lb
00QP01	Pitch Attitude	00QO01	deg
00QR01	Roll Attitude	00QR01	deg
00RP01	Pitch Rate	d/dt (00QP01)	deg/sec
00RR01	Roll Rate	d/dt (00QR01)	deg/sec
10DF01	F/A Cyclic Stock Position	10DF21	%
10DF02	Pedal Position	10DF22	%
10DL01	Lateral Cyclic Stock Position	10DL21	%
10DV01	Collective Position	10DV21	%
20MT51	#1 Rotor Mast Torque	(1)	in-lb
DF1001	F/A Cyclic Stick Rate	d/dt (10DF21)	%/sec
DF1002	Pedal Position Rate	d/dt (10DF22)	%/sec
DL1001	Lateral Cyclic Stick Rate	d/dt (10DL21)	%/sec
DV1001	Collective Position Rate	d/dt (10DV21)	%/sec
01AV50	C.G. Vertical Acceleration	01AV50	G

(1) Expression $= 14\,(15MT20 + 16MT20)\left(\dfrac{63025}{3.24\,(30RM03)}\right) - \left\{\dfrac{250100}{\left(\dfrac{10DF22}{2} + 50\right) + 24.75} + 49.20\left(\dfrac{10DF22}{2} + 50\right)\right\}$

Because the load level survey, like all helicopter operations, had many more undamaging than damaging cycles, a weight factor was added to the analysis. The goal of the weighting function was to force the regression to fit well to the damaging loads above the endurance limit while paying less attention to the low loads far below the endurance limit. A sigmoid function accomplished this task by stressing damaging to undamaging loads by a factor of 10 to 1. This resulted in equations that were more accurate above the endurance limit and less so below it. This is a desirable

attribute as non-damaging loads are irrelevant when calculating component life. The only condition that should be avoided is overpredicting the non-damaging loads to such an extent that they become damaging.

To guarantee conservatism, the equation must never underpredict loads. The correlation results are shown in Figures 14-18. These plots compare the predicted loads with the actual loads. Ideally, the correlation would be perfect, and all the data markers would lie on the diagonal line passing through the origin. But some points are below the line and in their case, the equations are underpredicting. The dotted line represents the 3σ offset. This line is 3 standard errors below the center line. Shifting the equation by 3σ guarantees that, given a normal distribution of data, the equation will only underpredict true loads less than 0.5% of the time. The underprediction on these few cases is greatly outweighed by the vast majority of the time that the equation is overpredicting. In fact, it can be seen in the regression plots that for all the data above the endurance limit, loads are overpredicted. Thus, the 3σ shift insures conservative damage calculations. The vertical distance between the center line and the 3σ offset line is the extra value that is added to the constant term in the regression equations. Notice the larger R^2 values result in smaller 3σ shifts. High R^2 values yield better fits, more accurate equations, and less standard error.

Because in most cases the datacodes from the trial helicopter were recorded in different units or on a different scale from that used for the original load level survey, a conversion process was necessary. Before the data could be entered into the regression equations, transform functions were applied. In addition, while all the control stick and attitude rates were recorded on the load level survey, they were not recorded on the trial helicopter. These were derived by taking the derivatives of control stick positions and attitude as a function of time. Main rotor mast torque was approximated by converting percent engine torques to total torque and subtracting losses to the tail rotor. The following 10 datacodes were required from the trial operator for regression analysis:

- Pitch attitude
- Roll attitude
- F/A cyclic stick position
- Pedal position
- Lateral cyclic stick position
- Collective stick position
- C.G. vertical acceleration
- Engine #1 torque
- Engine #2 torque
- Rotor RPM.

The last three datacodes were used only to calculate an approximate main rotor mast torque.

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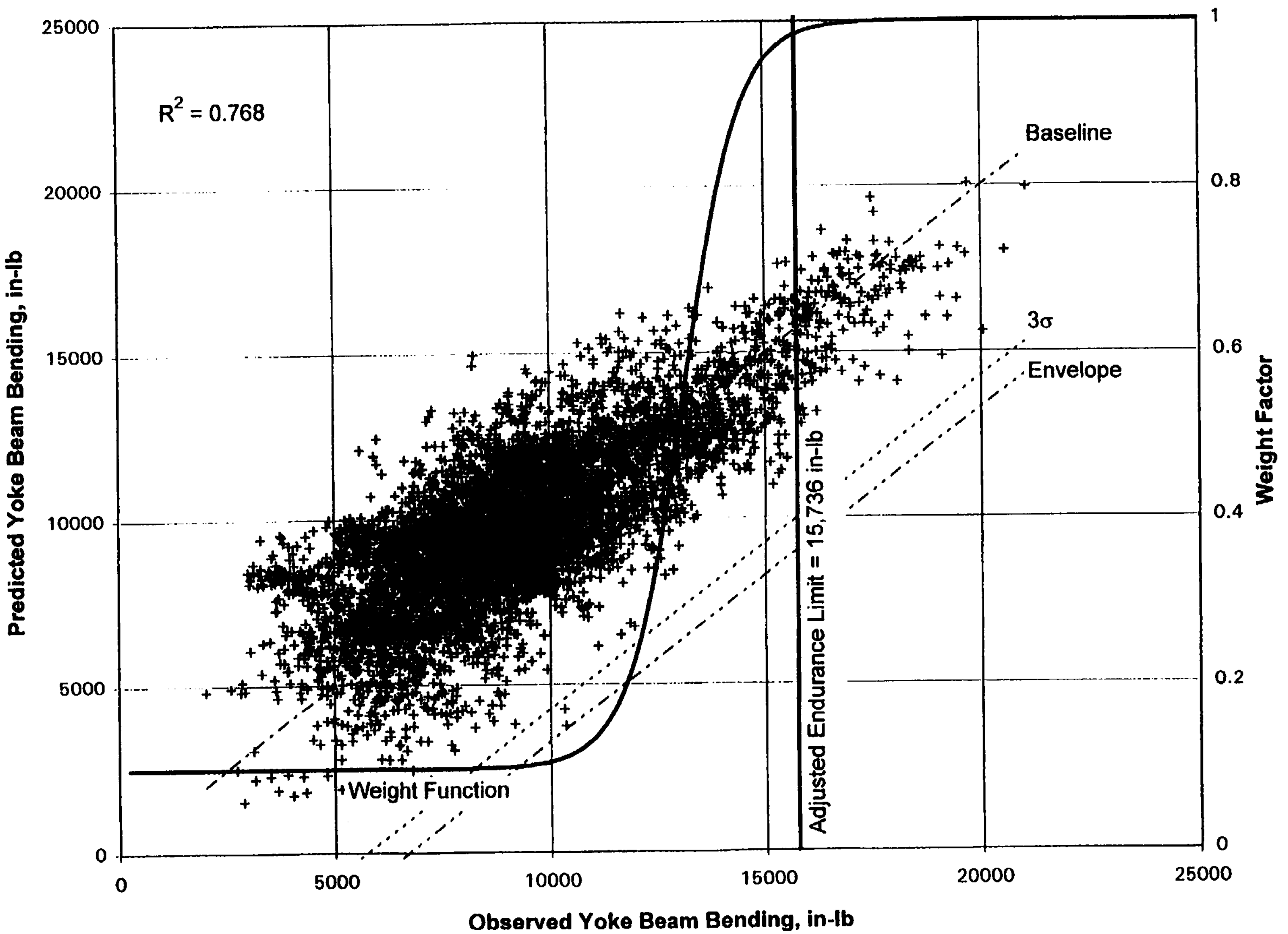

Figure 14. Correlation Plot, Oscillatory Loads for Main Rotor Yoke Assembly

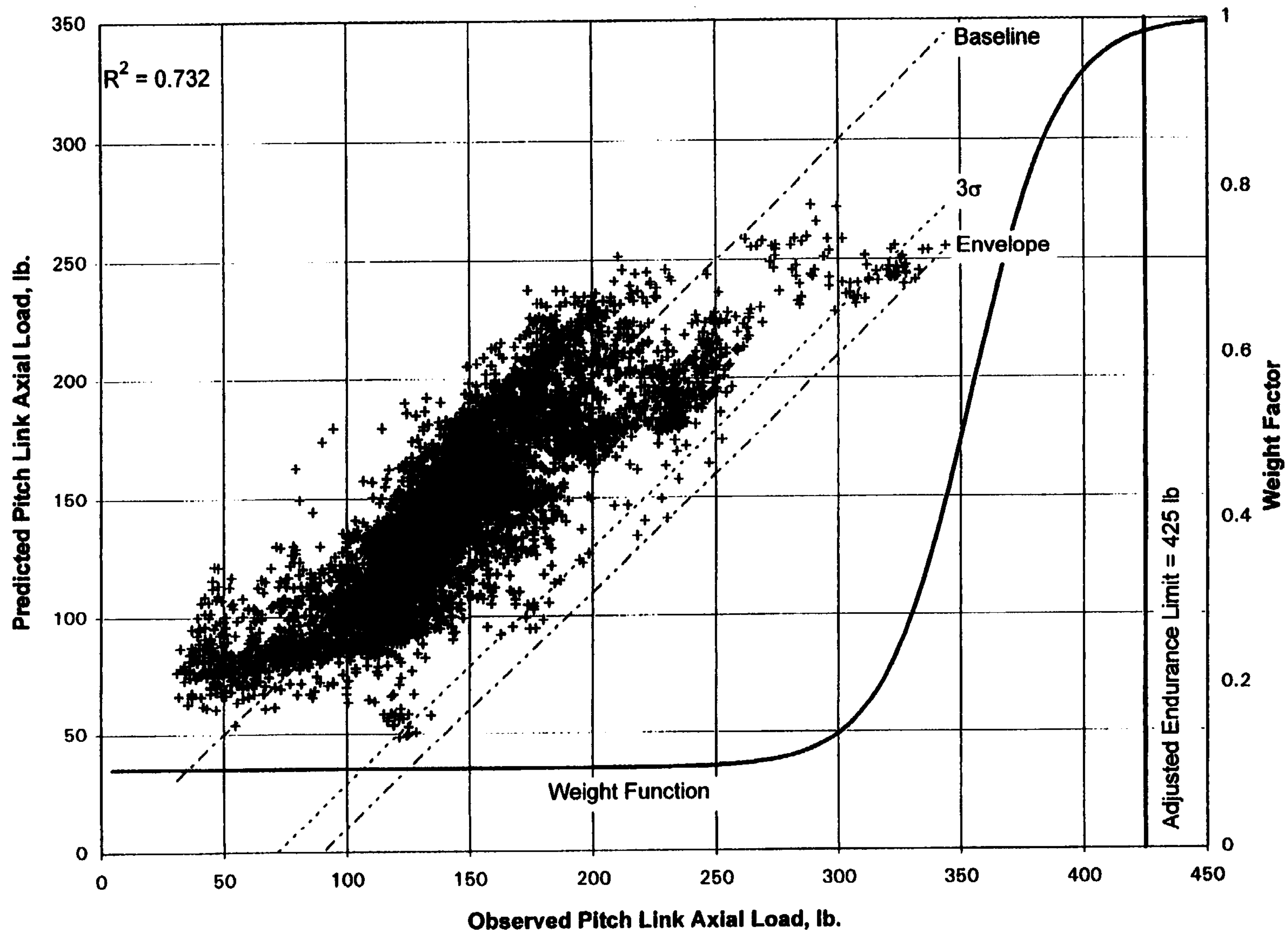

Figure 15. Correlation Plot, Oscillatory Loads for Spindle and Rephase Lever

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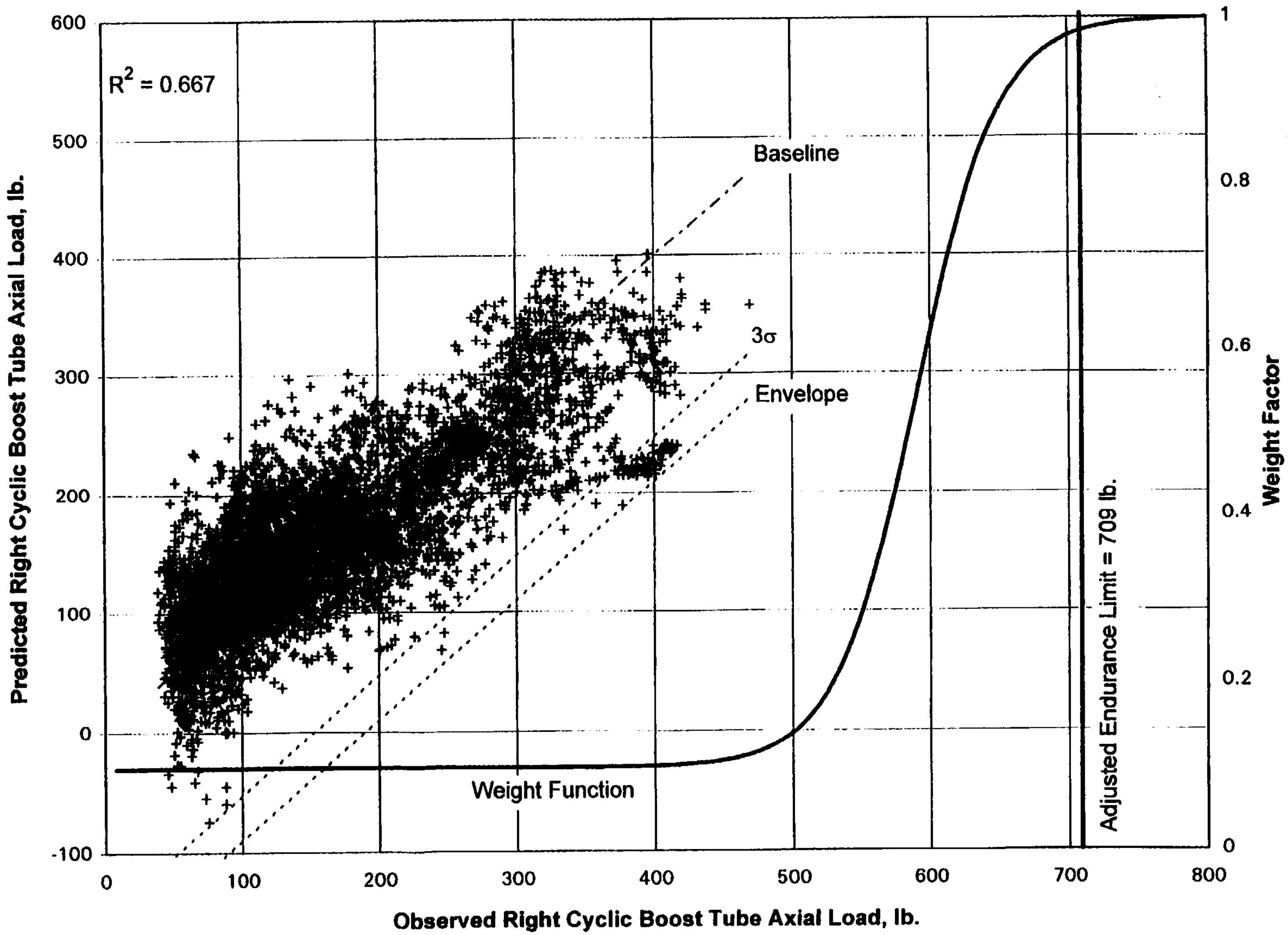

Figure 16. Correlation Plot, Oscillatory Loads for Swashplate Inner Ring

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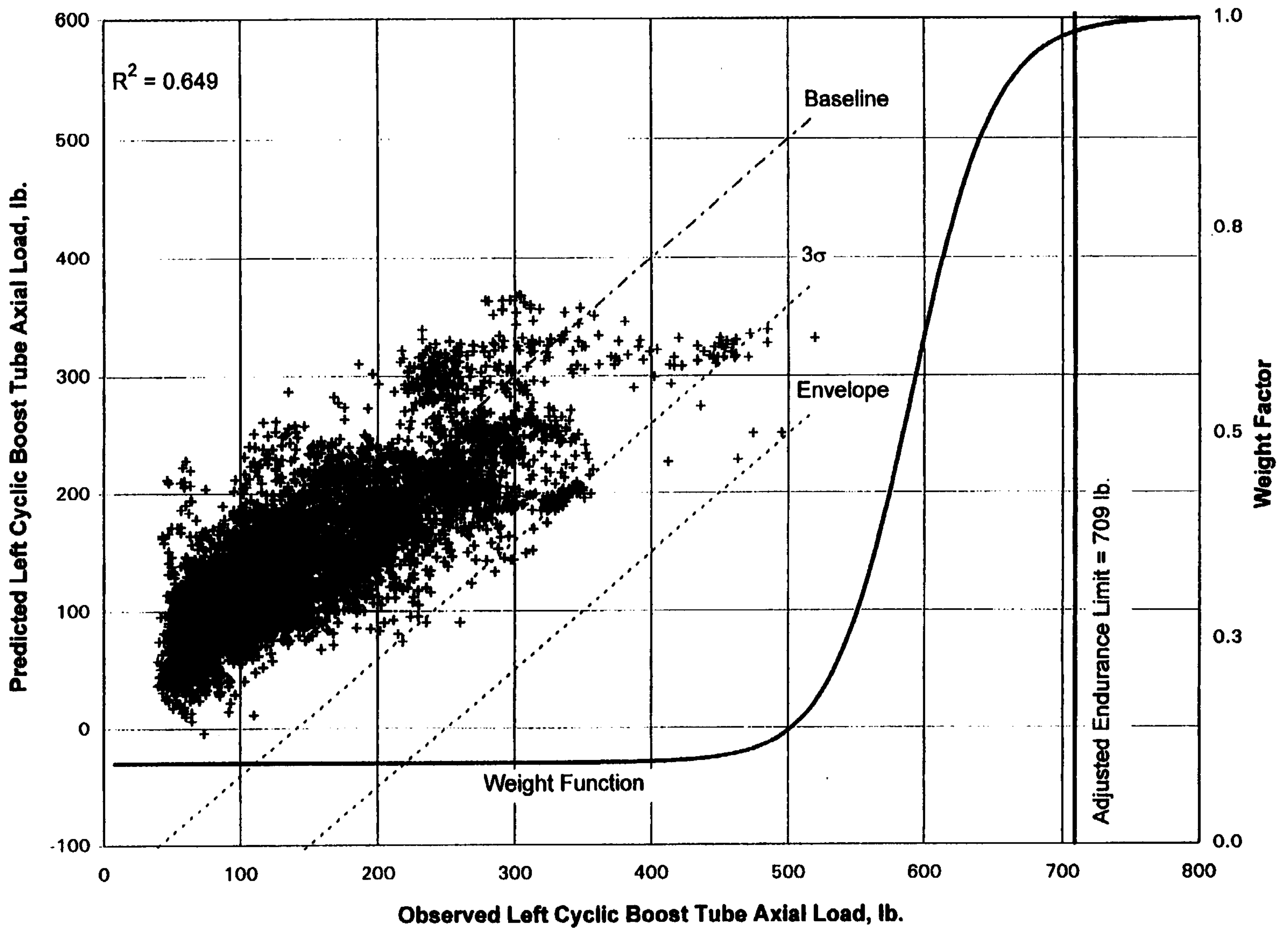

Figure 17. Correlation Plot, Oscillatory Loads for Left Cyclic Boost Tube

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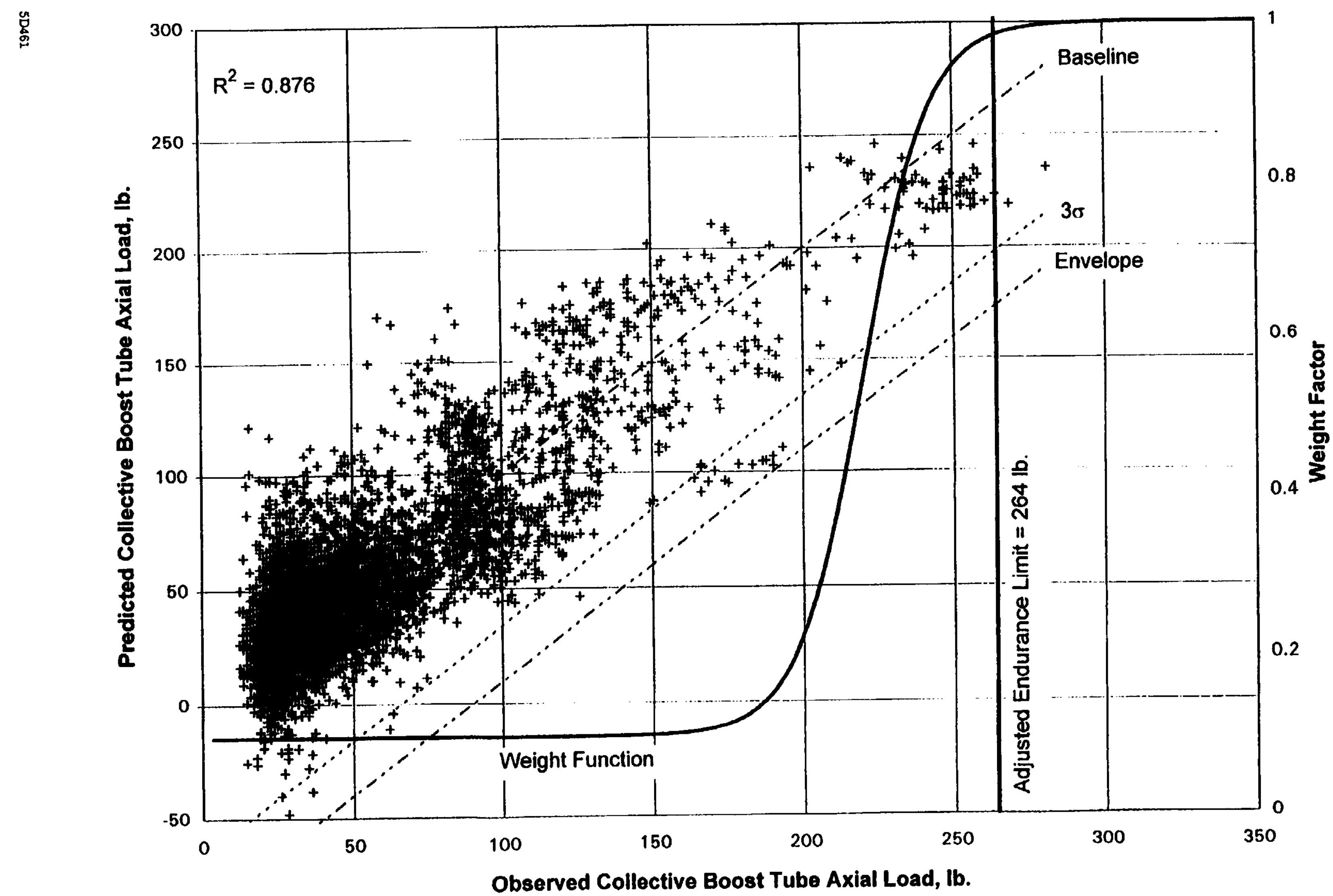

Figure 18. Correlation Plot, Oscillatory Loads for Collective Boost Tube

5.3 FLS RESULTS

The damage rates for FLS loads versus the current Safe-Life rates are shown in Figures 19-23. Swashplate inner ring and rephase lever assembly life predictions were longer than the Safe-Life baseline. The FLS main rotor yoke and collective lever were shorter than that of the baseline. The fin spar analysis was removed from consideration due to inadequate correlation.

To demonstrate the sensitivity of small shifts in the loads, damage predicted by FLS without the 3σ conservatism were also plotted. The FLS main rotor yoke and swashplate inner ring lives joined the rephasing lever in predicting longer lives than those predicted by the baseline and FCR. This shift had no noticeable effect on collective lever life.

FLS has the potential to predict longer component lives than that of FCR because FLS bypasses the spectrum concept, removing the additional conservatism inherent in the damage defined for each spectrum condition. However, FCR already shows significant extension of component lives and does not have as much trouble with voltage offsets as FLS does. Currently, the primary area where FLS excels is in the prediction of loads from undefined maneuvers. This potential benefit applies more to military applications, however, and not the benign, relatively predictable spectrum flown by most commercial users.

As described above, not all regression equations were successful in predicting longer lives. The FLS damage rate on the collective lever highlights a major problem with the FLS technique. The equations, by nature, are sensitive to changes in the input variables. A loss of any input nullifies the entire equation. Even more difficult to detect is a mean shift in one of the inputs. For example, in the longitudinal cyclic stick position gage, the post-testing measurement for a centered stick was -9%. With this offset, a stick position that should read 25% forward will erroneously read as 30%, when converted to the load level survey scale. This input variable will be skewed. When plugged into the regression equation, it will result in error. This error propagates into any terms containing longitudinal cyclic stick position and is magnified to a 20% error in the position squared term. In regression equations where longitudinal cyclic stick position is a major player, the size of the accumulating errors has the potential to drive normal cruising flight loads up above the endurance limit. With the equations predicting damaging loads for virtually all flight conditions, part lives drop off precipitously. In these cases, FLS will predict lives much shorter than those found through the current Safe-Life method.

An example of the sensitivity involved in FLS is shown in Figure 24. The longitudinal cyclic stick position sensor was erroneously converted to the load level survey scale without the necessary negative sign. All regression equations using this variable yielded damage rates far greater than the Safe-Life Method. In the example, one week of flying with the bad equation showed enough damage to suggest replacing

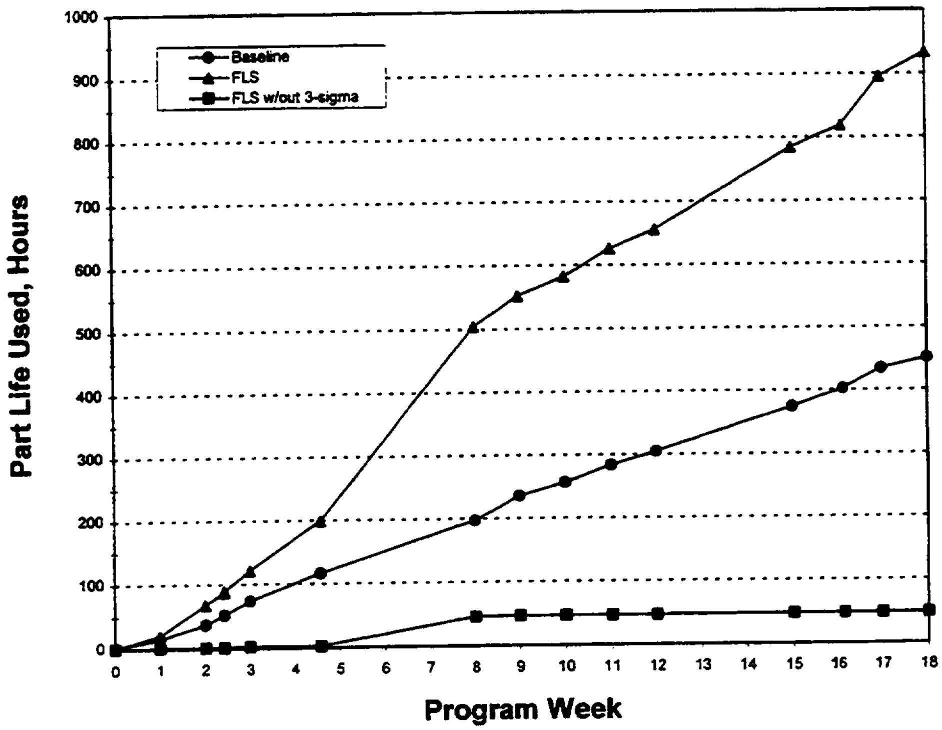

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Figure 19. Life Usage Comparison - Main Rotor Yoke

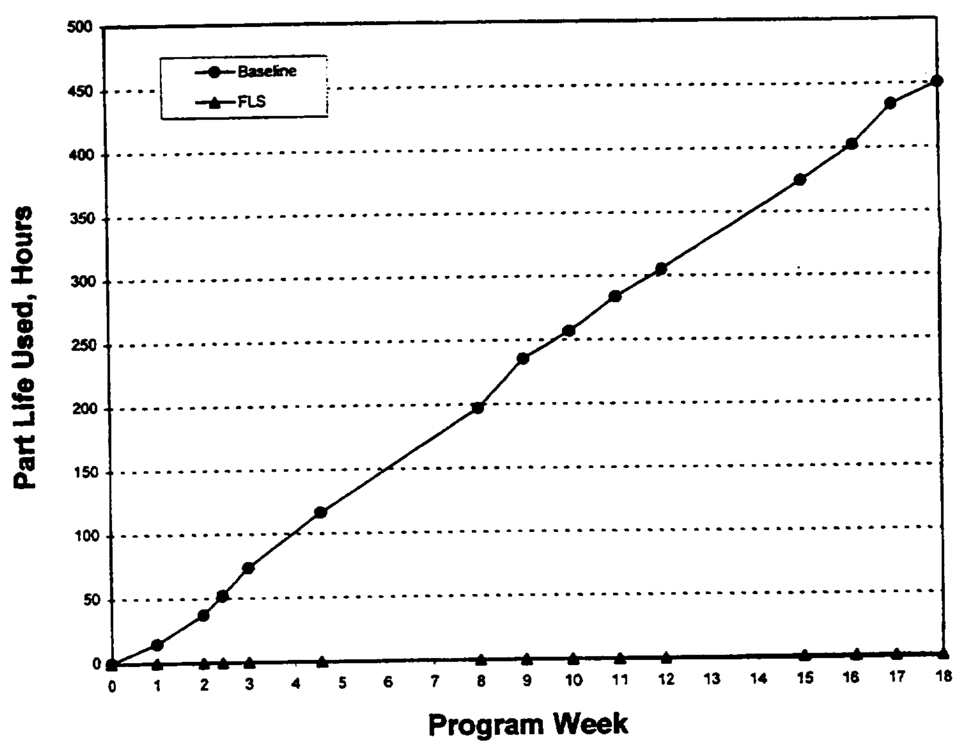

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Figure 20. Life Usage Comparison - Spindle

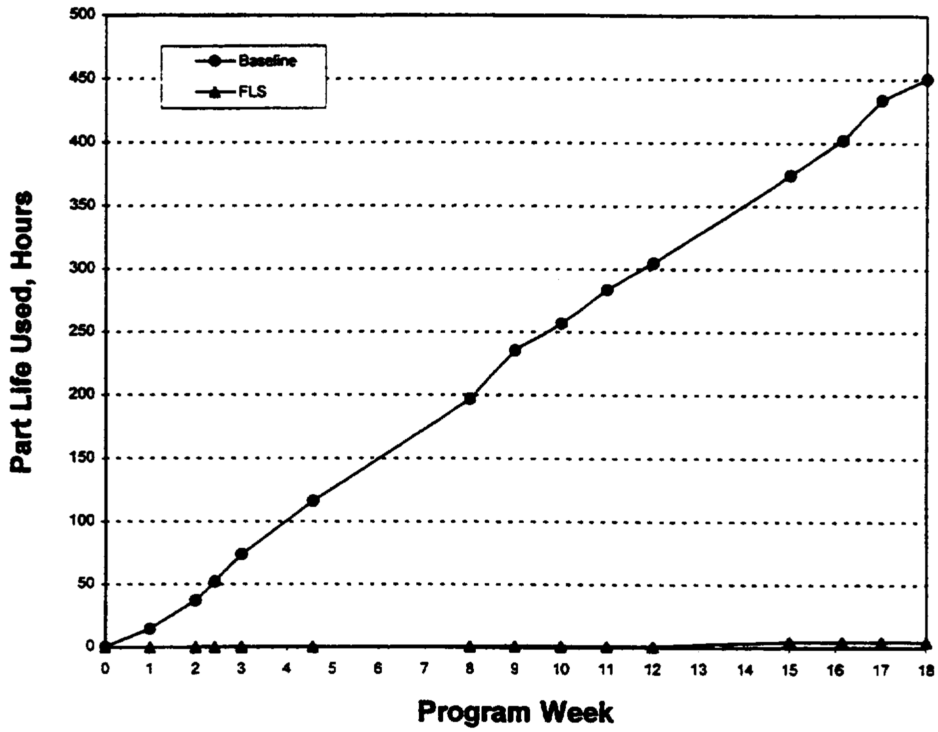

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Figure 21. Life Usage Comparison - Rephase Lever

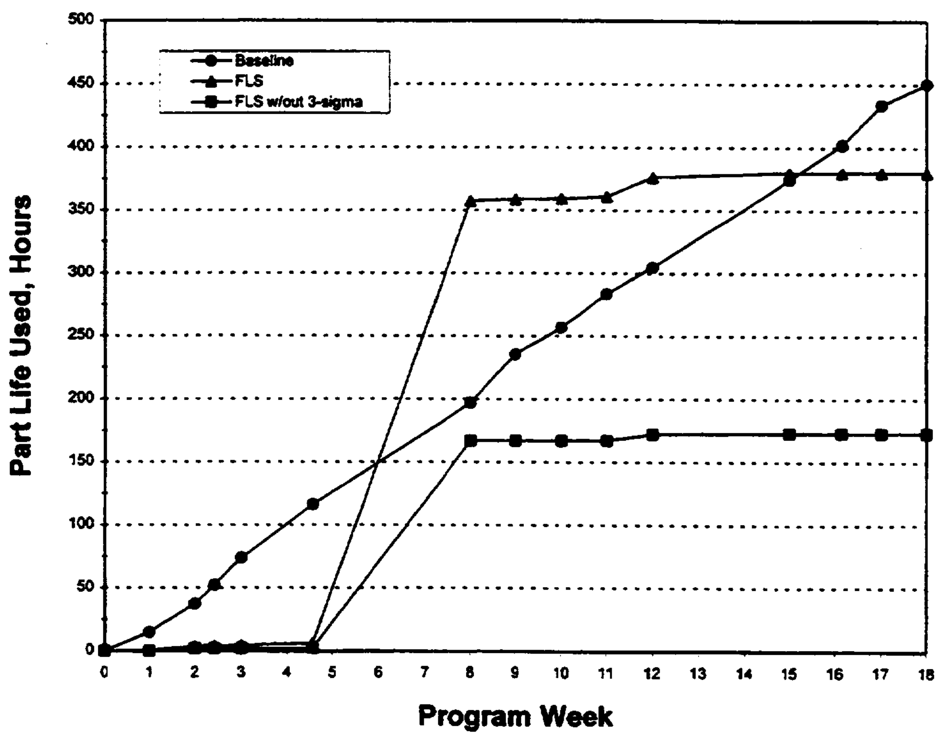

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Figure 22. Life Usage Comparison - Swashplate Inner Ring

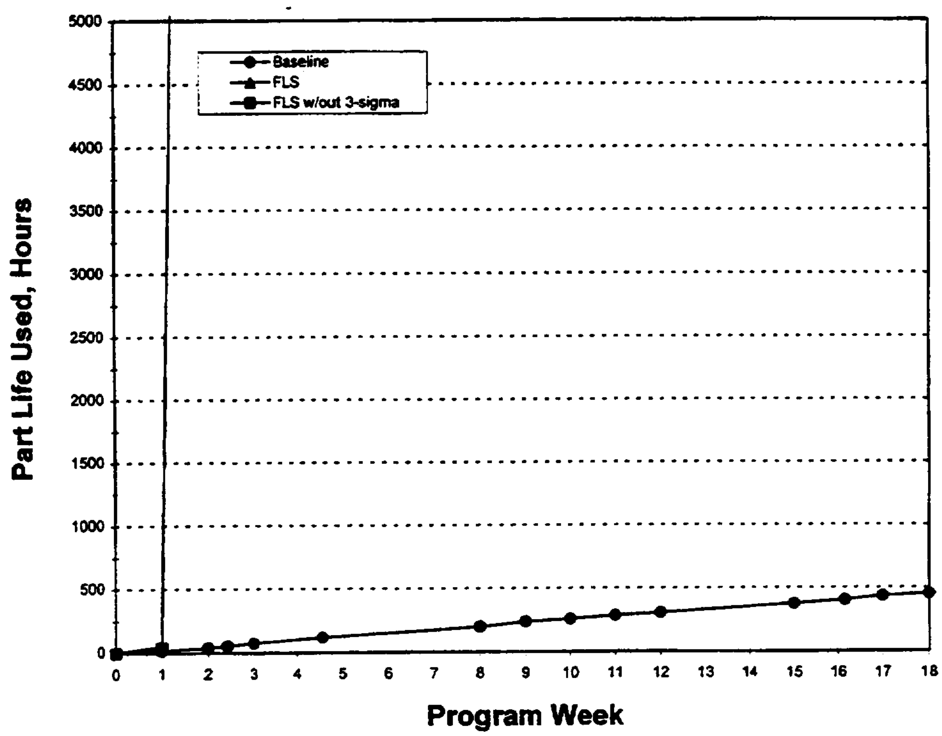

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Figure 23. Life Usage Comparison - Collective Lever

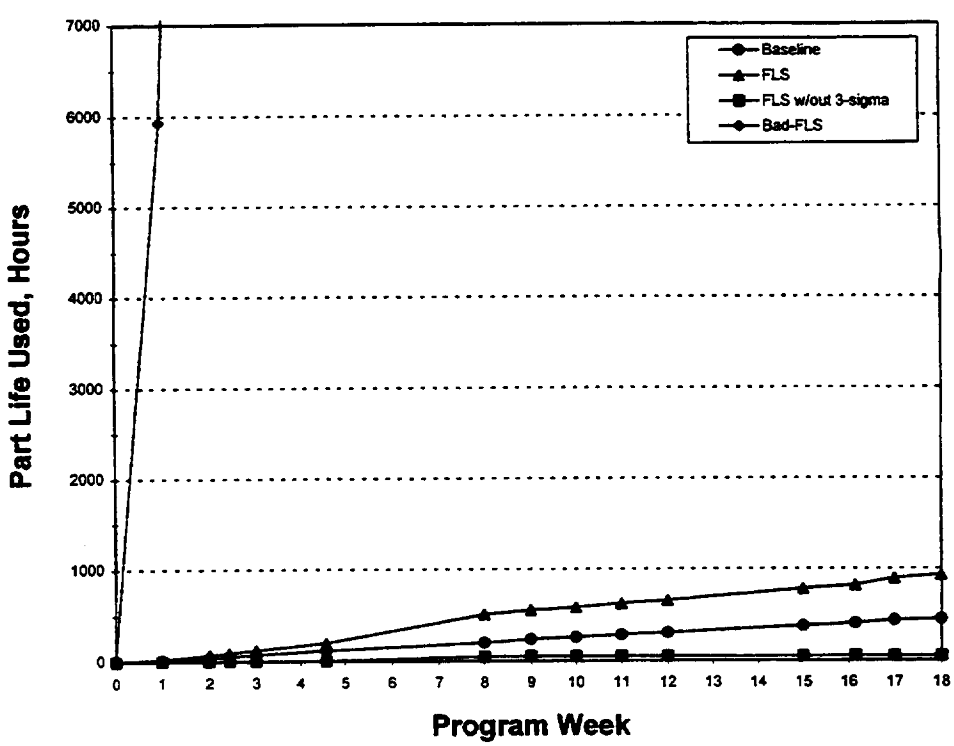

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Figure 24. Regression Equation Sensitivity - Main Rotor Yoke Damage from Correct Equation vs. Damage from Wrong Sign

the component. By nature, the equations are very sensitive to their inputs. FLS implementation will have to include periodic validity checks on all the sensors.

The sensitivity problem emphasizes the need for accurate, clean input data if FLS is to be implemented properly. Data spikes, bad gages, and mean data shifts all adversely affect the predicted loads. Another level of sensitivity is added to the problem by the flatness of the S-N curve in the high cycle region. Since the majority of flight loads are close to but under the endurance limit, a slight offset in the equations can turn the vast majority of non-damaging loads into damaging loads, ruining any chance of extending component lives.

FLS results would benefit greatly with a better database from which to develop the regression equations. Future load level surveys should include airspeed and gross weight as time-varying parameters. The insertion of these variables into new regression equations would greatly increase the correlation results, as well as produce acceptable correlations with fewer terms. This, in turn, would reduce the sensitivity of the equations to mean shifts in the data. In addition, cleaner data from the HUMS aircraft would also improve correlation. Any bad data point creates an outlier that skews the regression. The data spikes and hanger operations in the dataset of the demonstrator helicopter were difficult and time consuming to remove, and there is a strong possibility that some bad data escaped detection.

6. COMPARISON OF METHODOLOGIES

The opportunity to conduct an operational evaluation of the HUMS system is of utmost importance. Another important aspect of the evaluation was the opportunity to make a direct comparison of the FCR and FLS methodologies to the manufacturer's baseline loads and component lives. Because component loads were also measured during the HUMS trial, a direct comparison of these loads to loads predicted using the measured mission spectrum (FCR) with certification loads, the loads derived by FLS for the operator mission and the manufacturer's baseline certification spectrum and loads was possible. This comparison of the predicted versus actual values also gives an indication of how conservative or unconservative each method may be.

Figures 25 and 26 are comparisons of the oscillatory load distributions for the main rotor yoke, spindle, and rephase lever in terms of the appropriate load parameter for the FCR measured mission spectrum and certification loads, the loads synthesized using FLS and the certification spectrum and loads. Figures 27 and 28 are the same data for the swashplate inner ring and collective lever with the addition of the load distribution for the directly measured cyclic and collective boost loads.

In general, the distribution for the certification spectrum and loads are conservative in the part of the distribution above the stated endurance limit. This supports the contention that the certification spectrum is conservative and load survey maneuvers are flown more aggressively. In the case of Figures 27 and 28, the measured data indicates that the operator pilots are flying less aggressively than the pilots flew during the load level survey.

Figures 29 through 33 are composites plots of the part life consumptions for the FCR, FLS, and Baseline. These were presented separately in Sections 3 and 4. They are combined so that the comparison of data for all methods can be made more easily.

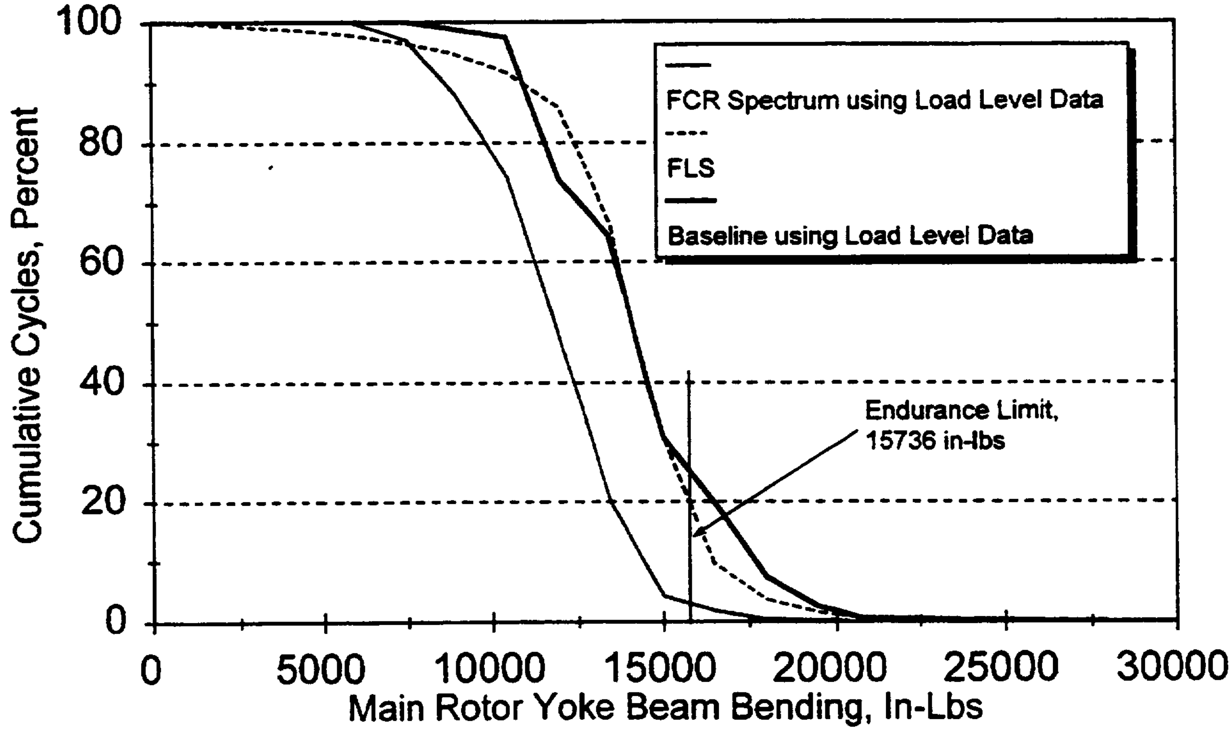

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Figure 25. Cumulative Cycles vs Oscillatory Load - Main Rotor Yoke

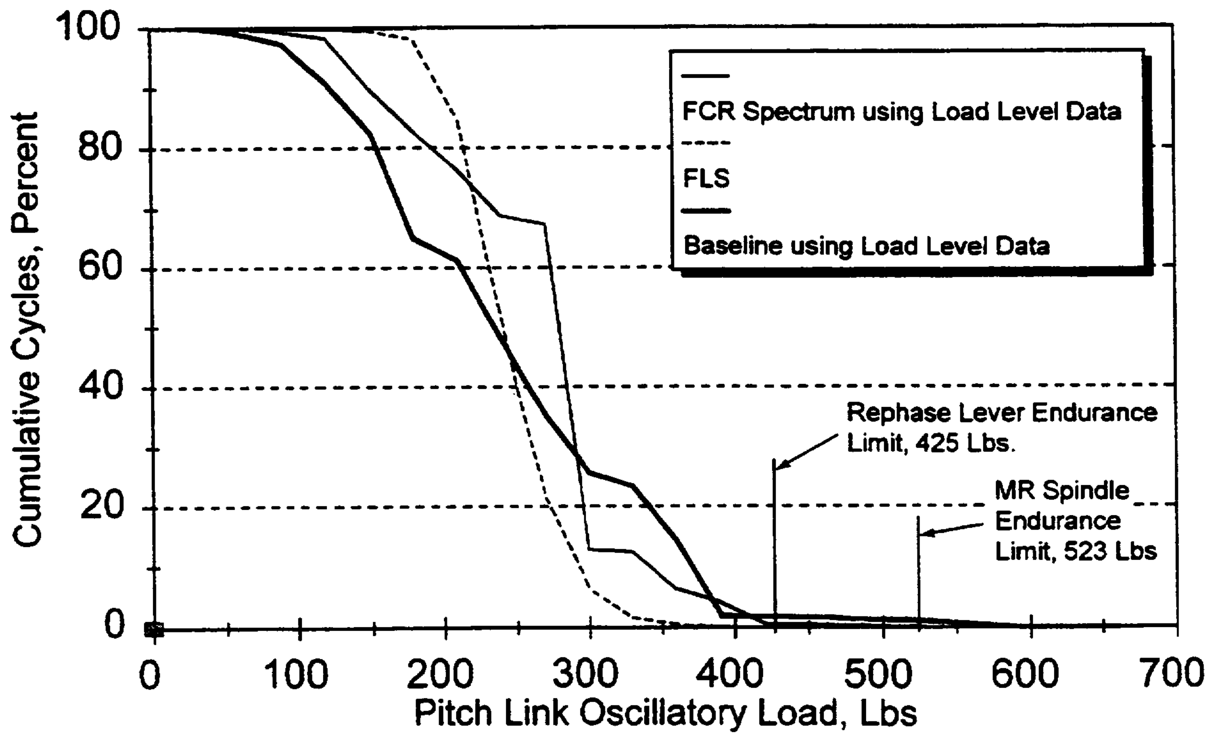

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Figure 26. Cumulative Cycles vs Oscillatory Load - Main Rotor Spindle, Rephase Lever

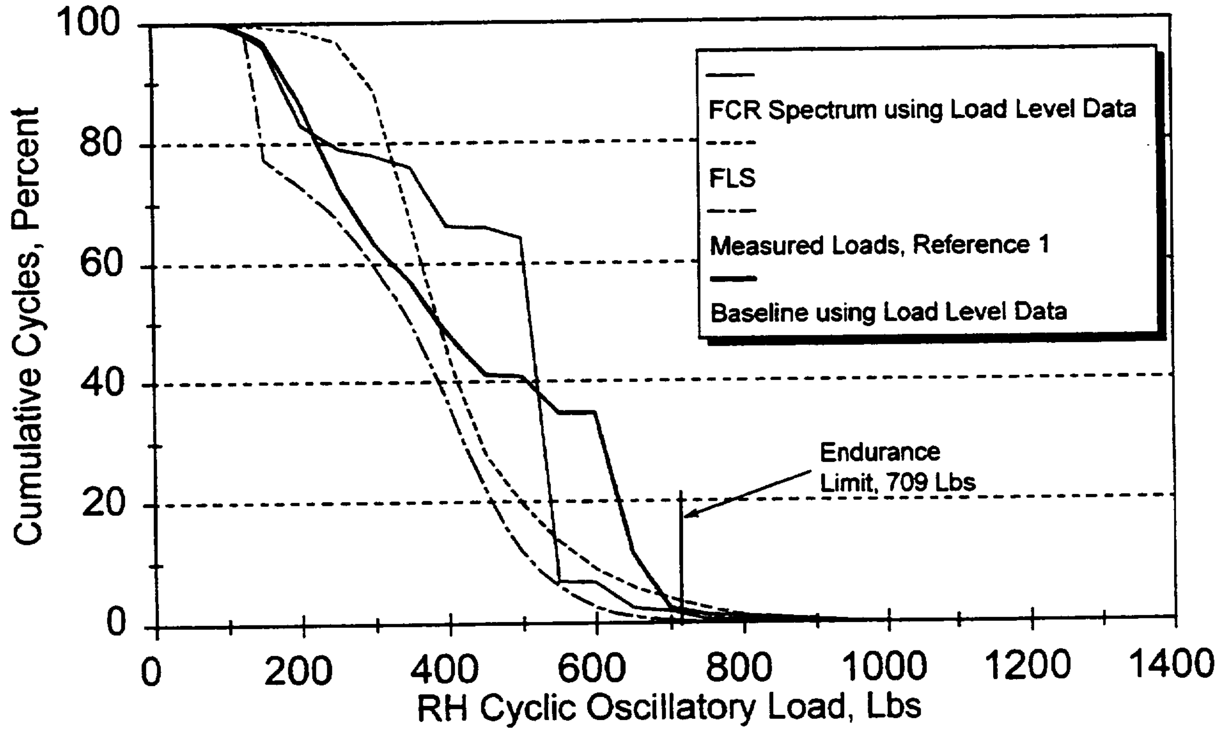

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Figure 27. Cumulative Cycles vs Oscillatory Load - Swashplate Inner Ring

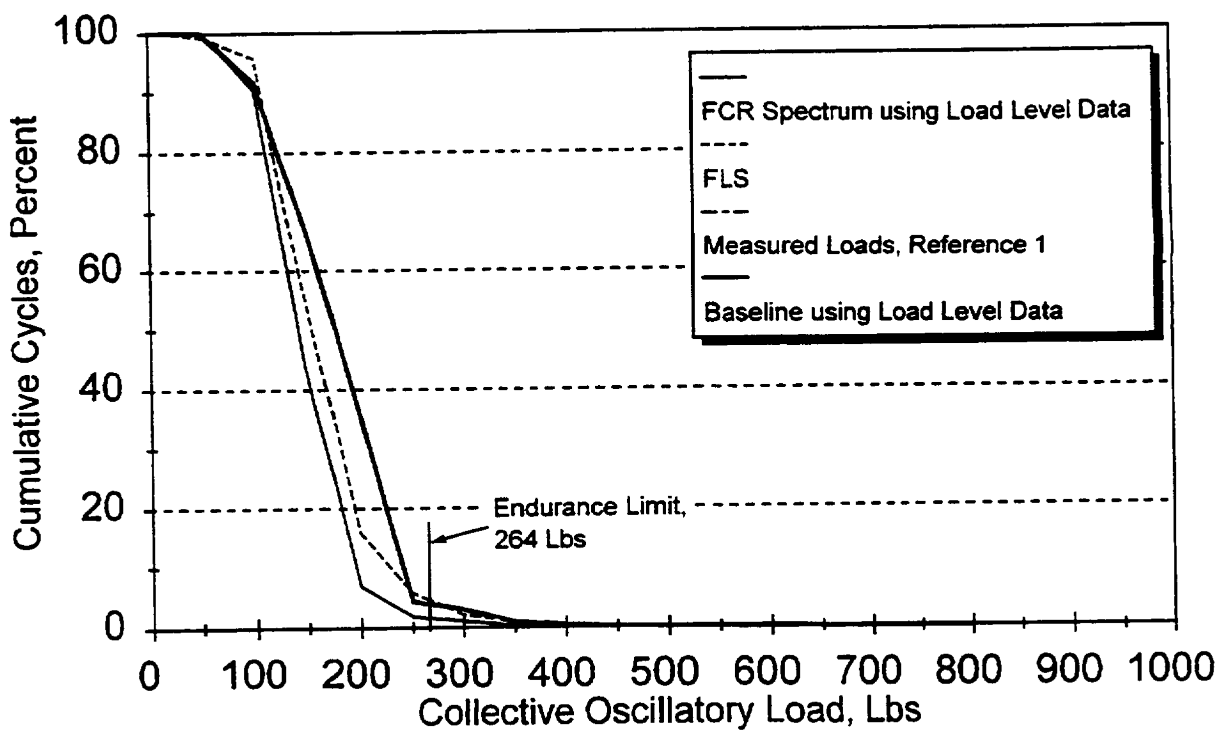

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Figure 28. Cumulative Cycles vs Oscillatory Load - Collective Lever

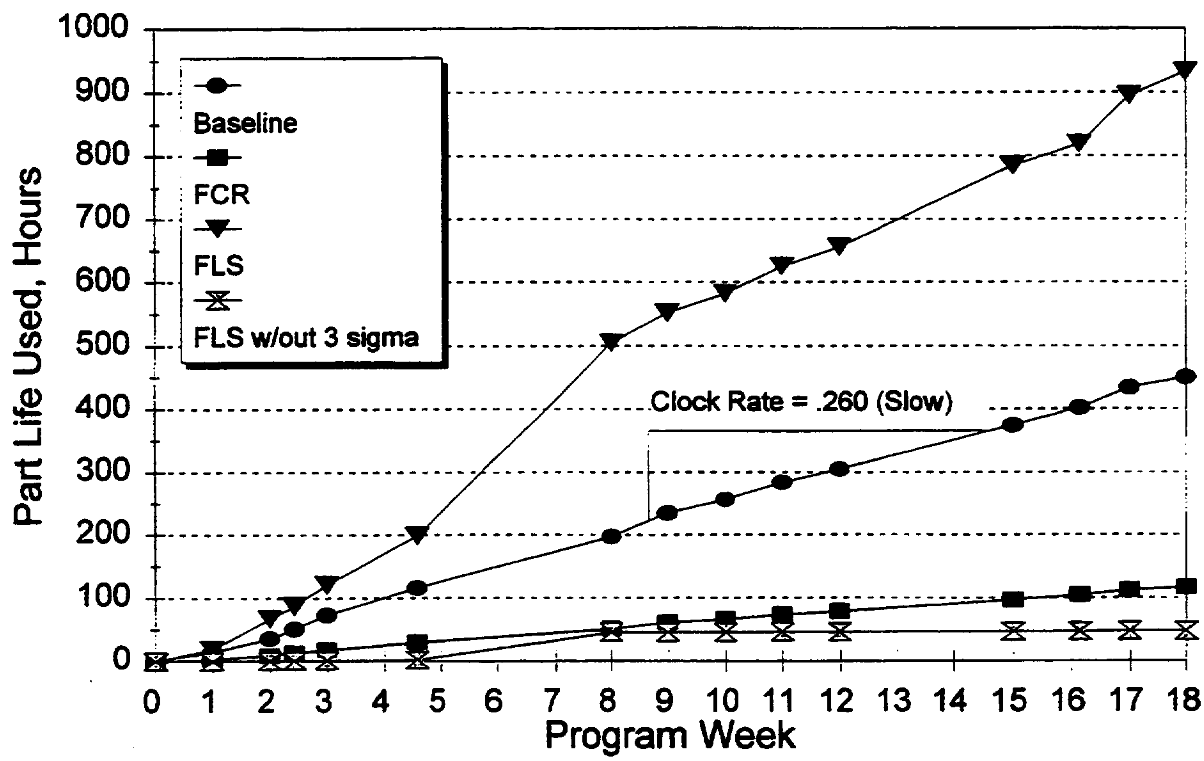

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Figure 29. Life Usage Comparison - Main Rotor Yoke

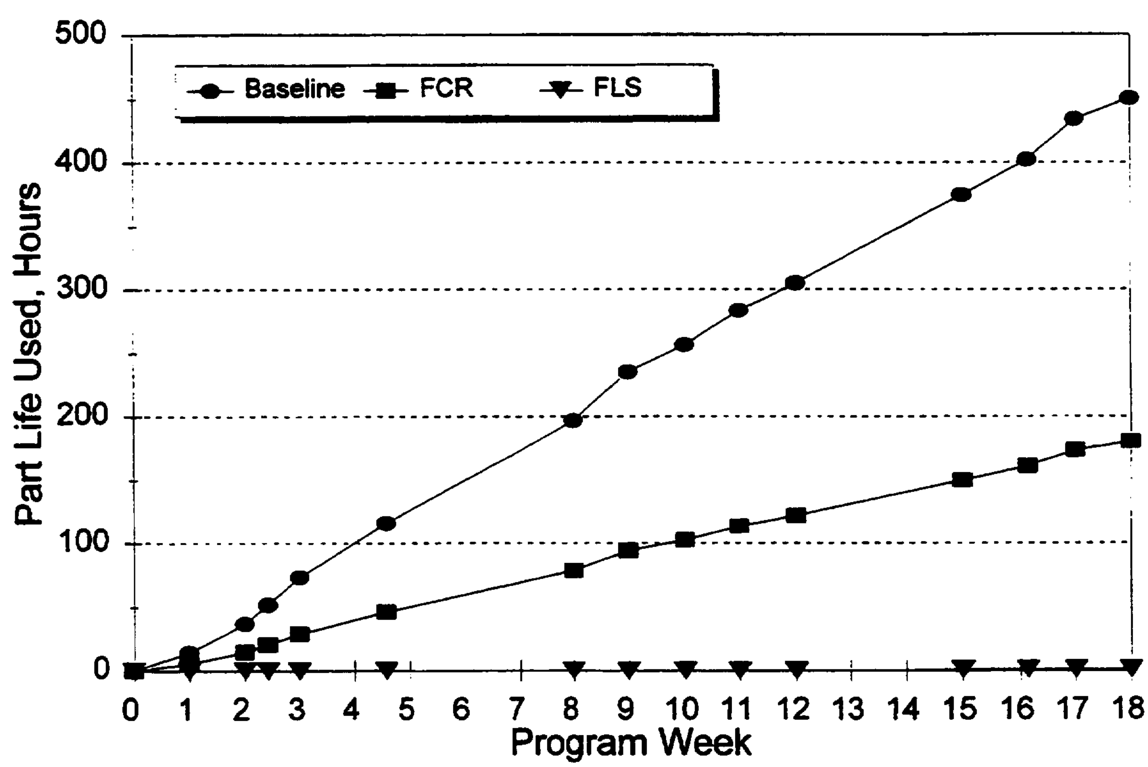

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Figure 30. Life Usage Comparison - Main Rotor Spindle

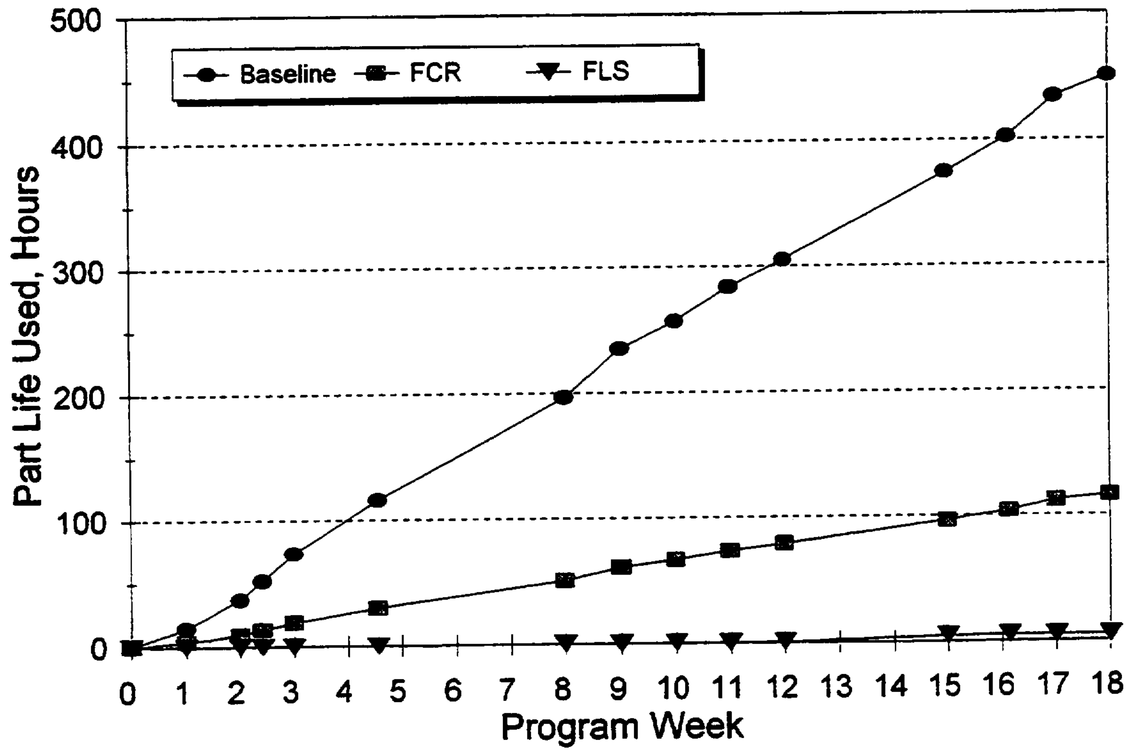

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Figure 31. Life Usage Comparison - Rephase Lever Assembly

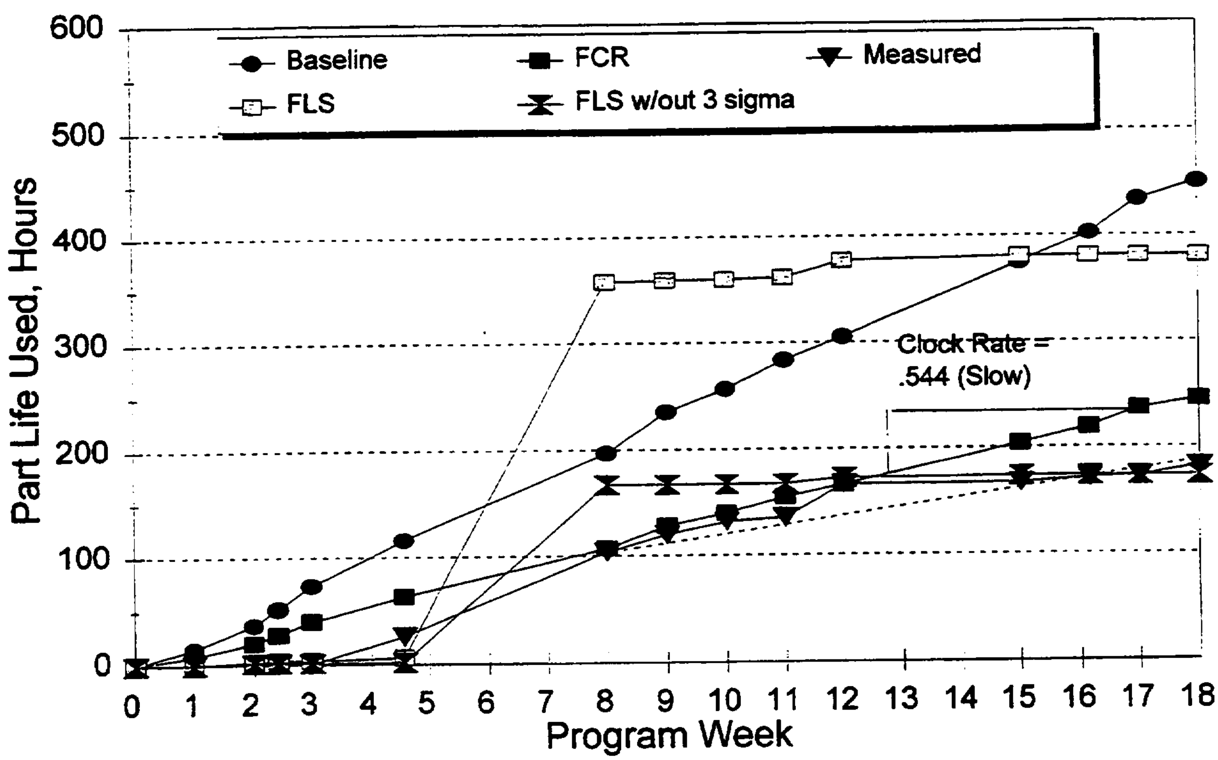

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Figure 32. Life Usage Comparison - Swashplate Inner Ring

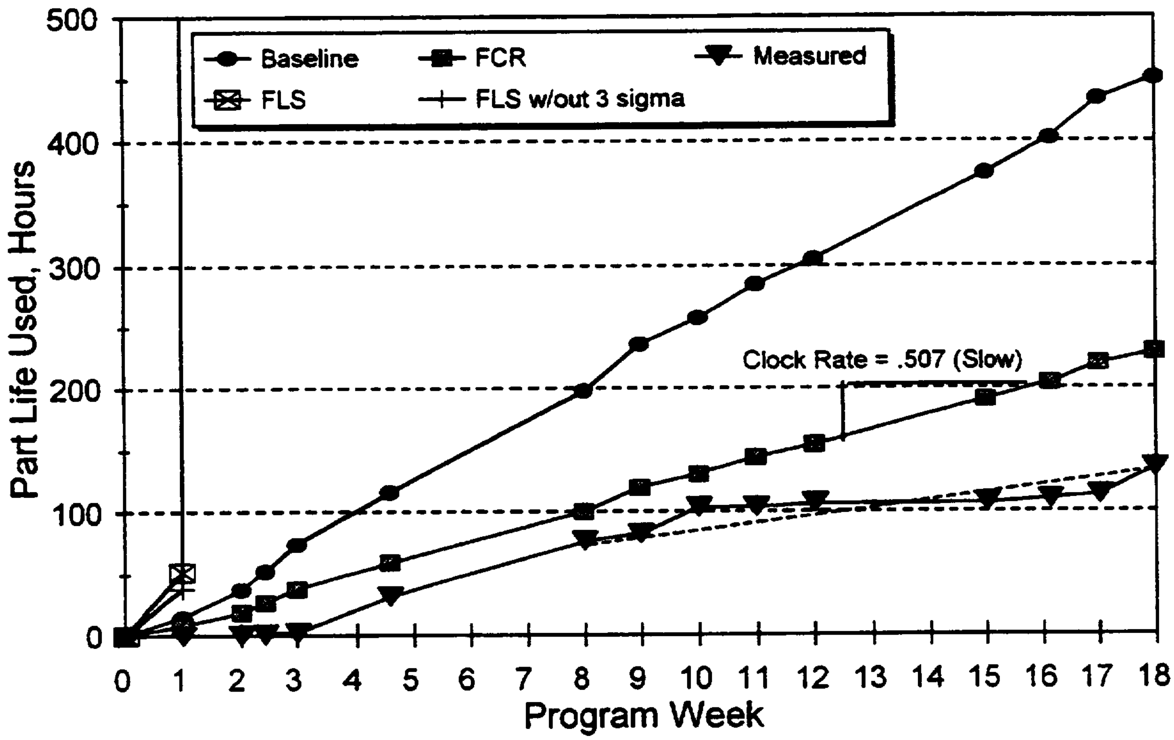

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Figure 33. Life Usage Comparison - Collective Lever

7. ECONOMIC IMPACT OF USAGE MONITORING

The use of HUMS must produce an economic benefit to the operator to help offset the cost of implementation and maintenance associated with such a system. In Reference 1, certain cost data was presented to emphasize this point. An example was presented where a 10% cost savings on selected life limited components was assumed for the helicopter used in this study, and the benefit was shown to pay for a $100,000.00 HUMS system in a matter of just 1,624 hours of operation. Figures were presented to support this determination. This same cost data is used with usage data from the HUMS trial results to determine the potential savings for the offshore oil support mission for these same components, as well as other components in the rotor and control system of the the helicopter used in this study.

To summarize some of the cost data:

- Total cost per flight hour is $615.89

  Parts replacement cost is $254.82
  Labor cost is $42.94
  Fuel/lube, power plant cost is $318.13

- Cost to "zero out" including components/overhauls/inspections is $1,036,017.00

- Cost of hub parts based on 5,000 hours of operation is $221,891.08

- Cost to replace main rotor yokes/spindles is $148,522.22

  Two yoke assemblies are $69,932.10
  Four spindle assemblies are $85,590.12

Usage monitoring via HUMS should have an effect on component replacement cost. This effect will be investigated for the study helicopter components using the FCR method together with the HUMS usage data. Table 12 is a listing of the components evaluated in this effort together with the projected component flight time when based on the HUMS usage data. For purposes of this illustration and to make a direct comparison to the original operator data, the main rotor yoke and spindle will be used for cost determination.

Table 12. **Summary of Flight Hours, Baseline vs. HUMS (FCR)**

Component	Baseline Retirement Life ~ Hr	Logged Hr	FCR Hr	% Life Increase	Projected Retirement Life ~ Hr
M/R Yoke	5,000	450	117	285	19,250
M/R Spindle	10,000	450	180	150	25,000
Rephase Lever	5,000	450	116	288	19,400
S/P Inner Ring	10,000	450	245	84	18,400
Coll Lever	10,000	450	228	97	19,700
M/R Mast *	60,000 RIN	1113 RIN	823 RIN	35	81,000 RIN
M/R Spline Plate *	60,000 RIN	1113 RIN	823 RIN	35	81,000 RIN

* Retired at 10,000 hours or 60,000 RIN. RIN is "Retirement Index Number" determined for low cycle fatigue (GAG) via the rain flow algorithm. Uses combined engine torque.

- Cost determination for the main rotor yoke

  Hourly cost - $12.59 (baseline) vs. $3.27 (HUMS)
  Cost/5,000 hr - $62,932.10 (baseline) vs. $16,346.00 (HUMS)

- Projected savings in 5,000 hours

  $9.32/hr or $46,600.00

- Cost determination for the main rotor spindles

  Hourly cost - $8.56 (baseline) vs. $3.42 (HUMS)
  Cost/5,000 hr - $42,795.00 (baseline) vs. $17,100 (HUMS)

- Projected savings in 5,000 hours

  $5.14/hr or $25,695.00

In summary, the cost on a 5,000 hour basis would be $72,300.00 with HUMS vs. $105,727.10 currently. This equates to a 33% savings in cost to the operator. If similar savings could be projected for all life limited components on the study helicopter, the savings would be very significant to the operator.

Additionally, there are spinoff or intangible costs savings not reflected in these cost savings determinations. There is a potential to reduce the labor costs when parts remain in service longer. The number of overhauls will be extended and the shipping and handling costs associated with parts replenishment will be reduced. The operator will also be better able to forecast spare parts requirements and thus better budget for spare parts costs.

8. CONCLUSIONS

A number of significant conclusions can be drawn from the study conducted. These are listed below:

1. There is a significant cost savings using FCR for the participating operator and any other operator performing a similar mission type.

2. The FCR technique can be implemented almost immediately with the following modifications:

    a. Implement the improved GW/CG measurement system (see Section 9, Recommendations).

    b. Use a pilot keyboard entry system, and use this value in conjunction with the fuel burn algorithm to predict gross weight if modification (a) is not implemented.

3. The measured oscillatory loads recorded during the HUMS trial and the predicted oscillatory loads from the HUMS FCR agree very well. This would indicate that the spectrum predicted from the HUMS FCR is valid for the mission that was flown.

4. The manufacturer's baseline (certification) oscillatory load distributions for the cyclic and collective boost loads are conservative when compared to the distributions predicted by HUMS FCR. This validates that the spectrum used to originally certify the study helicopter is conservative, as required by Federal Regulation.

5. The Flight Load Synthesis (FLS) approach requires more refinement before implementation. Areas requiring improvement include:

    a. Better methods to predict scatter factors for the correlation.

    b. More correlation data from higher magnitude flight loads.

    c. Periodic check on the validity of key aircraft parameters.

    d. Possible incorporation of FCR to identify flight regimes before applying FLS.

    e. The quality of FLS was adversely affected due to:

        (1) Lack of GW and airspeed data for the certification data used to develop the correlation equation coefficients.

(2) Electrical drift or loss of calibration reference during the HUMS trial for several of the aircraft parameters resulting in poor correlation.

In summary, the HUMS trial has significantly expanded the scope of usage monitoring. The knowledge gained provides all parties—operator, manufacturer, equipment supplier, and the certification authorities—with a better understanding of the advantages of usage monitoring. It has also revealed some areas where improvement must be made. The improvements which have been identified are technologically possible to achieve. They can be infused into the next generation HUMS system easily and will result in a much improved usage monitoring system.

9. RECOMMENDATIONS

As a result of the HUMS trial, several recommendations are being made to further improve the usage aspect of monitoring. These are:

1. Refinement of Gross Weight/C.G. measurement system to account for wind/RPM/collective influences.

   a. Use data to refine GW algorithms and evaluate the effect on system accuracy.

   b. Explore other electro-mechanical approaches to accomplish better accuracy.

   c. For near term implementation of usage, provide pilot keyboard entry of GW.

2. In conjunction with the program outlined in Item 1 above, install a mast moment device (no slip ring required) as a means to improve correlation of main rotor and controls for FLS. This device has been flown at BHTI with good success.

3. Provide cockpit display of usage information including but not limited to:

   a. Gross Weight and C.G.

   b. Flight time (cumulative)

   c. RIN value

4. Install promising devices from 1 through 3 above on the helicopter used in this study for operational trial.

5. Devise a method to automatically perform a periodic check on the validity of key aircraft parameters such as:

   a. Control positions

   b. Ship attitude (pitch, roll, etc.)

   c. Load factor, N_Z

   d. Airspeed

6. Use an existing load level survey, which has a more robust suite of parameters, to refine FLS techniques and improve the quality of the correlation as indicated by R^2.

7. Explore improvements to usage monitoring with a hybrid of FCR and FLS. FCR would be used to identify flight regime, level flight, turns, climb, etc. It would be possible then to derive more specific equation coefficients with much better R^2 values. This should result in better overall correlation for the FLS technique.

8. Define the architecture for a production usage system and define how it should be integrated into the operator maintenance procedures.

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13. ABSTRACT ***(Maximum 200 words)***
The objective of this study was to evaluate two techniques, Flight Condition Recognition (FCR) and Flight Load Synthesis (FLS), for usage monitoring and assess the potential benefits of extending the retirement intervals of life-limited components, thus reducing the operator's maintenance and replacement costs. Both techniques involve indirect determination of loads using measured flight parameters and subsequent fatigue analysis to calculate the life expended on the life-limited components. To assess the potential benefit of usage monitoring, the two usage techniques were compared to current methods of component retirement. In addition, comparisons were made with direct load measurements to assess the accuracy of the two techniques. The data that was used for the evaluation of the usage monitoring techniques was collected under an independent HUMS Flight trial program, using a commercially available HUMS and data recording system. The usage data collect from the HUMS trial aircraft was analyzed off-line using PC-based software that included the FCR and FLS techniques. In the future, if the technique prove feasible, usage monitoring would be incorporated into the onboard HUMS. The benefit of usage monitoring was identified under work accomplished during the first phase of this activity. The results from the operator's perspective is presented in the report NASA CR198446 (ARL–CR–289; DOT/FAA/AR–95/50). For the selected dynamic components analyzed, the results of the evaluation of the FCR and FLS techniques indicate a potential for extending retirement lives. This is due to the damage accumulation rate for the FCR and FLS techniques being slower ("slow clock") than the current method using actual flight hours as the basis for retirement times. Of course, the benefits of usage monitoring are dependent on how the aircraft is operated. Based on the mission flown for this aircraft, which is flying work crews to offshore oil platforms, the flight hours charged against retirement times could be reduced by 50% or greater. Thus, the operator would gain a considerable payback in reduced maintenance costs due to extension of retirement intervals. The FCR technique, which only modifies the helicopter maneuver spectrum relative to the manufacturer's baseline, was considered more practical and lower risk to implement compared to the FLS technique. However, the FLS technique could be refined to overcome shortcomings found.

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